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PROJECT STUDY OF POSTAL MISSILES

Translation of "Avant Projet d'Engin Postal"
Engins Matra and Giravions Dorand, January 1962

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PROJECT STUDY OF POSTAL MISSILES

This note was written in collaboration between:

- ENGINS MATRA
- GIRAVIONS DORAND

GENERAL CONTENTS

CHAPTER 1

INTRODUCTION

1 - 1 - HISTORY

1 - 2 - PURPOSE OF THE PRESENTATION

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1 - 1 - HISTORY

At the beginning of its activity in the field of special missiles the Engins Matra Company had established at the request of the Service Technique Aeronautique a project study of a postal missile corresponding to an over-all program with the following data:

- Range . . . . 1500 km
- Commercial payload . . . . 200 kg

Take-off and landing had to be performed on a site of restricted dimensions.

This program was abandoned in favor of studies of military ground/air and air/air missiles, and the project study was not further pursued.

Nevertheless, initial thoughts led to certain basic ideas which were patented in France and in several foreign countries. In particular, after several years of investigation, the essential claims were granted in the U.S.A. in 1954. They appeared to offer even today certain interest in their application to missiles for conveying loads to be delivered with a certain precision at some predetermined area of restricted dimensions (for example a site of some tens of meters each way), which can be located in the neighborhood of population centers.

The essential basic idea* consists of making use of a rotary wing with its axis of rotation lying in the longitudinal axis of the missile.

The machine takes off vertically with retracted blades. The "return to ground" maneuver comprises a descent of the missile towards the landing point, followed by a recovery at low altitude, which finishes up in a vertical climb at the end of which, when the speed is sufficiently low, the rotor is brought into action.

Descent takes place in autorotation, an automatic pilot stabilizes the missile by controlling the rotor. Guidance by remote control or automatic guidance by a vertical beam permits the positioning of the missile with good precision on its landing field.

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*French Patent No. 933, 558 dated 10, 9, 46.
U.S.A. Patents No. 2, 684, 213 dated 10, 7, 54 and 2, 886, 261 dated 12, 5, 59.
British Patent No. 69, 947 dated 12, 6, 50.

In the original patent, the blades were folded along the body inside channels provided for this purpose. This led either to an increase in the drag, or to difficulties in the arrangements of the propulsion units and equipment. In addition, this arrangement made the unfolding of the blades a delicate operation.

At this point the Engins-Matra Company took note with interest of the proposals of a foreign inventor, Mr. V. Isacco, who made a controllable parachute rotor in England for airborne troops. Mr. V. Isacco, known before the last world war for his helicopter work, considered his parachute as the final aim. He wanted to settle down in France for personal reasons.

The Engins-Matra Company thought that Mr. Isacco's idea could profitably complement those which it intended to use in its postal missile. The parachute could be a first stage, so to speak a mock-up which was capable of further extrapolation.

Mr. Isacco, in his parachute rotor, used two blades, each consisting of a set of telescopic elements, which came into action under the combined effect of small powder charges and centrifugal force. When compressed and folded, the rotor was hardly more bulky than a conventional parachute. When extended, its diameter reached 5.60 m, and the assembly weighed 15 kg.

The Matra-Company, after studying the prototype made in England, decided to build five machines. The tests of the first four, carried out at Bretigny, finished up in failure. The fifth rotor, launched from a helicopter over the Cormeille region, functioned properly, and a film shows clearly the unfolding of the blades, and the descent to the ground. This film still exists and can be examined.

It must be emphasized that, in the Isacco parachute, the extension of the blades, each consisting of nine elements, has never been the cause of failure. The uncontrolled rotor did not allow an autorotative descent with sufficient stability. It is the dynamic instability and faulty setting of the blades which were the reasons for the four failures of Mr. Isacco.

Thus, it was not until the fifth attempt, when, launched from a hovering helicopter and after a modification of the automatic device for changing the blade pitch, the Isacco parachute carried out a correct descent.

However, official scepticism over the interest in such a parachute on the one hand, and the lack of a program which envisaged an application other than that foreseen by the inventor on the other hand, did not permit us to pursue the development of Isacco's work.

Several months ago a patent was published in the United States covering a rotor with extending blades, which resembles in its broad lines the Isacco rotor, and which offers in its application a certain similarity with the Robert and Henault (Matra) Patents.

It should be noted that this concerns a rotor intended for the recovery of the cabin of a space missile (Von Saurma Patent, No. 2,969,211 dated 24/1/1961). The Saurma blades have flexible skin and are made to extend by the pressure of compressed air. The position of the Von Saurma rotor in relation to the missile axis and its means of control are related in a certain degree to the Robert Patents.

The American patentee, Von Saurma, mentions moreover, Robert and Isacco (U.S.A. Patents). It should be thought, therefore, that the claims granted to Saurma are limited as a result of the precedance of Isacco on the one hand and Robert on the other (concerning the latter, see the U.S.A. claims contained in Appendix 4).

1 - 2 - PURPOSE OF THE PRESENTATION

We have described above the past contribution of the Matra Company to the postal missile problem. Apart from the construction and tests of the Isacco parachute, this contribution has been limited to analytical studies. In fact, at the time when the project studies of the postal missile were completed, the technology was not yet sufficiently well mastered to permit approaching the construction of such a missile.

This is no longer true. It is at present possible to make use of missile hardware of great reliability, if this requirement is taken into account in the design stage. Such reliable material would permit the postal missile to fly over inhabited regions without excessive risks.

In addition, the Engins Matra Company has acquired considerable experience in the field of ground/air missiles. In particular, the Company has designed, constructed ~~and tested~~ R-422 and R-431 missiles, of which 200 specimens have been launched.

However, not being specialized in the technique of rotary wings, the Engins-Matra Company has sought the collaboration of qualified engineers. Thus, the Company proposed to the Giravions Dorand Company to take over the responsibility for the design and construction of the rotor. The Giravions Dorand Company, after examining the problem, has agreed to put its experience at the disposal of the project and offer its guarantee. In fact, the Dorand sustaining unit design which we present in the present submission substantially differs from the Isacco rotor. It is distinguished by a greater simplicity of its conception, and for this reason should offer a greater reliability in use. In addition, the Giravions Dorand Company will make a great contribution to the study of the handling and guidance of the missile with rotor unfolded. Having regard to these diverse contributions, we thought it worth while to present the project studies of a postal missile. The program selected is less ambitious than that which we recalled at the beginning of the preceding Section. The program permits nevertheless, and this is essential, a feasibility study of the formula. We have adopted the following type of mission:

- Range: .... 300 km
- Payload .... 50 kg

Our purpose is to show that the landing solution which we propose is already capable of realization and is well adapted to the postal missile problem, having regard to the degree of recovery which it makes possible, and the precision of landing which it assures. These two factors enter directly into the profitability of the operation.

Other factors must be taken into account before arriving at an operational version of the missle, particularly the specification of the ground installations which leads to the minimum cost of operation, and the integration of the ground equipment within that of general air navigation systems.

We believe that these problems could be the theme of a further study, and we submit a project of an experimental postal missile whose guidance is provided by well-tested and available ground equipment, (operated by radar and remote control), but which could be made lighter in the operational stage.

1 - 3 - CONTENTS OF THE SUBMISSION DOCUMENT

In Chapter 2 we present in summary form the general properties of the missile and of the guidance equipment used. This is followed by the operational version envisaged by us and its possible further developments.

Chapter 3 contains the planning of the design, construction and tests of the missile. This planning is foreseen over a period of four years. The financial forecast of the cost of the study and the sequence of payments are then established.

The specification of the missile is given in Chapter 4, namely aerodynamic configuration, propulsion unit, arrangement and handling of the load. A detailed description of the rotor is given. The performance of the missile is determined.

In Chapter 5, we deal with the guidance of the missile during the different phases of its flight. We explain the chosen working principles, the guidance laws envisaged, and the equipment to be used.

Three technical appendices complete this chapter:

- Appendix I gives the reference frames used in the course of guidance, namely reference frames related to earth and reference frames related to the missile.

- Appendix II deals with the analog computer used for the descent and approach phases.

- Appendix III shows the piloting devices which control the flight of the missile, both during the cruising phase and the descent with the rotor unfolded.

- Finally, in Appendix IV, a copy of the patents granted in the U.S.A. covering the recovery of the missile by means of a rotor is reproduced.

CHAPTER 2

GENERAL CHARACTERISTICS

2 - 1 - THE MISSILE

2 - 2 - THE GUIDANCE

2 - 3 - OPERATIONAL VERSION

2 - 1 - THE MISSILE

The performance values selected for the cruising flight are:

Altitude Z = 20,000 meters

Speed M = 2

The following considerations lead to these two values:

- A high flight altitude permits, on the one hand, the flight of the missile outside the zone of air traffic and, on the other hand, facilitates the operation of the electromagnetic equipment necessary to provide the guidance in cruising. (This latter factor will be the more effective the greater the lift required of the missile).

- While for a range of 300 km the speed of the missile does not count greatly, for much longer missions it is appropriate to make use of a supersonic missile.

To fit these values the best propulsion method is the ramjet, both because of its simplicity and its low weight without fuel. In addition, the selection of the ramjet covers the possible extension of the cruising flight (Z = 25,000 meters, M = 3, for example).

A first stage of propulsion is necessary to permit a rapid attainment of near-cruising speed by the missile. This booster is a solid propergol (isolane) propulsion unit, the casing of which remains on board the missile after burning out.

The powder booster is placed in the missile fuselage, while two ramjets are attached to this fuselage.

The rotor has two blades. Each blade consists of three elements, of which the last is telescopic. The rotor diameter is 6 meters.

The wing system is of cruciform layout. Each of the wings is provided at its extremity with a leg, forming thus the support polygon of the missile on touching the ground.

The take-off weight of the missile is 1170 kg, and the landing weight is 590 kg. Its length is 5.70 m and its width, 3.20 m (retracted rotor).

The take-off of the missile is vertical. Traversing the 300 km is accomplished in 580 seconds. The descent with rotor unfolded which brings the missile down to the ground from an altitude of about 1000 meters vertically above the landing point takes 80 seconds.

Complete mission of the missile thus takes approximately 11 minutes.

2 - 2 - THE GUIDANCE

The ground equipment for guidance which permits the missile to follow this trajectory consists of a guidance control unit installed near the landing point. No special equipment is needed at the launching point. Since the take-off is vertical, the missile reaches rapidly an altitude of 7000 meters after which it is in optical contact with the guidance control unit and can therefore be taken over by the unit. Until this instant, the flight can be controlled entirely by the automatic pilot.

In order to provide guidance during cruising, the guidance control unit includes the following equipment:

- An LY radar, modified so as to provide the localizing of the missile all along its trajectory, provided the missile is equipped with a responder.

- An analog computer comprising standard computing elements, which, starting from the position of the missile, computes the guidance signals.

- A remote control system which transmits the computed signals to the missile.

Since LY radar does not provide a sufficient precision of localization to bring the missile to a landing site of 100 m^2, it is necessary to replace LY radar, to be called here the "cruising radar", by more precise localization equipment for the approach and descent phases.

The guidance control unit equipment will therefore include a conventional approach (total) radar which takes over the missile at about 20 km from the arrival point. By setting up the telemetering equipment of this radar at the center of the approach site, it is possible to achieve at the end of the descent a precision within the limits of scatter, which must be ensured in landing.

The entire equipment used for guidance, including the ground equipment and missile-borne equipment, does not require special development. It has already been tested and proved in many applications. In particular, so far as we are concerned, the guidance equipment constitutes the pre-guidance apparatus of the ground/air missile R-422.

2 - 3 - OPERATIONAL VERSION

Concerning the missile, we have already indicated above that the concept submitted here appears to us to be well adapted to operational utilization, owing to its great precision in landing and the high rate of recovery of the equipment. It is hardly appropriate at the moment to go further and envisage an operational version of the missile without a precise knowledge of the program and conditions of utilization.

On the other hand, concerning the guidance system, it should be noted that the installation of two radars at each end of a two-way communication link constitutes a large operational control net. Devices which may lead to a more modest ground equipment system can be envisaged.

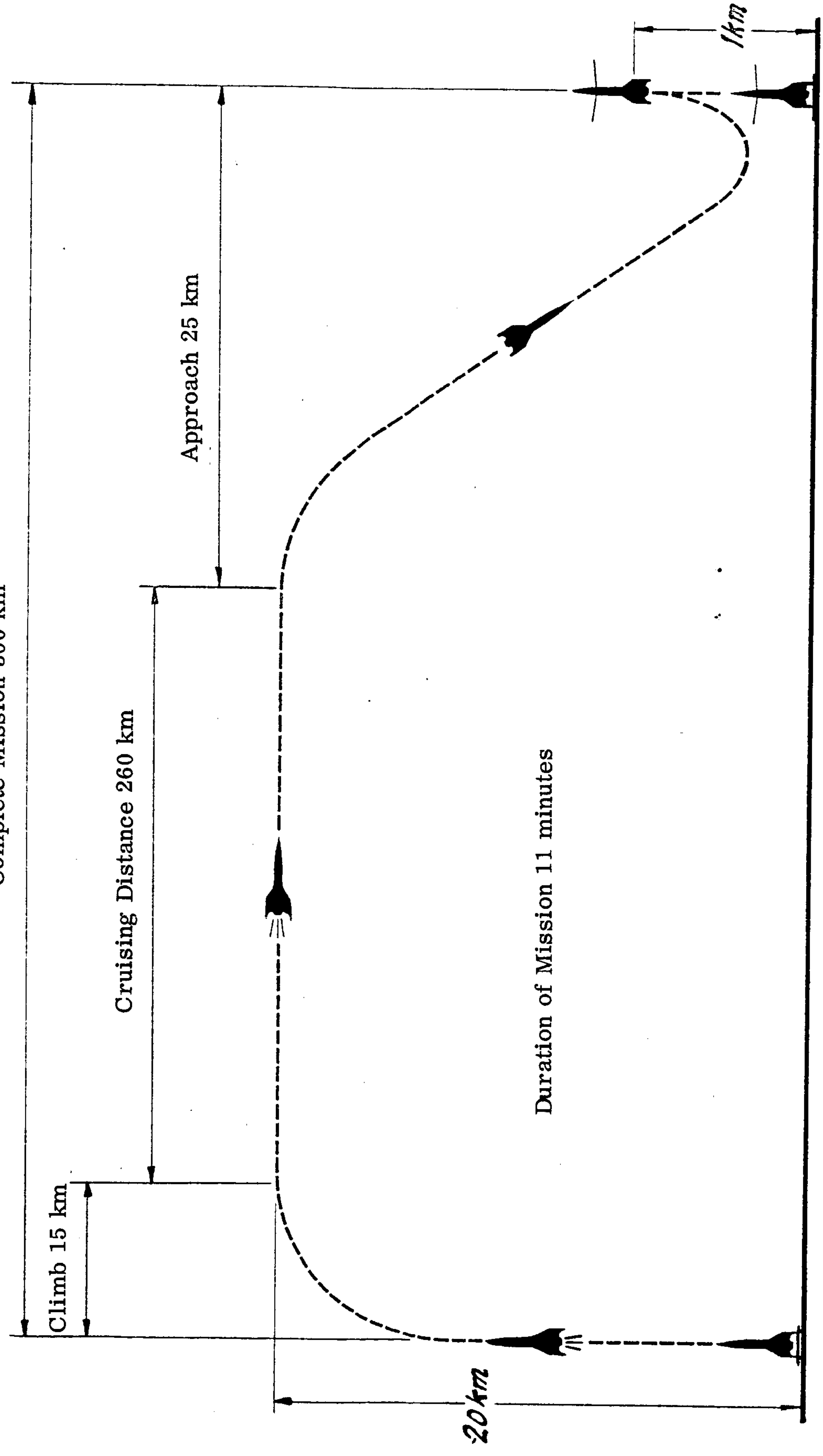
Thus, cruising can be directed fully automatically by a missile equipped with an altimeter probe and navigation equipment (VOR receiver, for example, of which the transmitting beacon is placed near the landing site). The missile carries out homing on the landing point during constant altitude flight.

The approach and descent with rotor unfolded can be carried out with lighter equipment. In particular, the following possibilities can be considered:

- Guidance by automatic remote control by using a localizer better fitted to the task than the approach radar (Cotal). In addition, it is possible to use the outgoing beam of this electromagnetic link for transmitting to the missile information issued by the computer, so as to assemble in the same apparatus the localizing and control signal transmitting functions.

- On-beam guidance by producing a mobile electromagnetic beam, the movement of which is programed so as to cause the missile to perform the approach and descent maneuovers.

The development of such specific equipment for the postal rocket program can be pursued only within the framework of an operational definition of the mission. In addition, it would have to be perfectly matched to the different phases of the missile flight, and particularly to the descent with rotor unfolded. An analysis and preliminary tests of this descent are essential, even if it is not necessary to reach the final aim of the development, in order to define the foundation for an operational version.


1km
Approach 25 km
Complete Mission 300 km
Cruising Distance 260 km
Duration of Mission 11 minutes
Climb 15 km
20km

CHAPTER 3

PLANNING AND PRICE

3 - 1 - PLANNING OF THE DEVELOPMENT OF THE MISSILE

3 - 2 - FINANCIAL ESTIMATE

3 - 1 - PLANNING OF THE DEVELOPMENT OF THE MISSILE

The planning of the design studies, construction and tests of the postal missile is presented on page 26. This planning extends over four years.

In order to establish the planning sequence we have defined a certain number of headings essential to the project, and we present the progress of work under these headings during four years. These headings encompass the general studies, the development and manufacture of components and the flight tests.

Each of these headings is associated with an index number which is also used in the financial estimate of the cost of this research.

We give below in chronological order a more detailed description of the work to be carried out.

First Year:

During the first year the design and manufacture of prototypes only will be pursued. In particular:

- The technical specification of flight control and guidance equipment are evolved.

- The analysis and wind tunnel tests of the missile are performed.

- The design of flight control and guidance equipment are started.

In fact, in considering this design work we have assumed merely adaptation of existing equipment backed by a great deal of experience. The performance which we expect from this equipment shouldnot require the development of new concepts.

- The project work, the completion of the type record, and the manufacture of two prototypes of the rotor, are carried out during the same period.

Second Year:

The following work will be carried out during the year:

- Design work on the adaptation of flight control and guidance equipment to the missile is pursued.

- Prototypes of the flight control and guidance equipment are type-tested in ground tests, subject to vibrations, accelerations, etc. In addition, guidance equipment is tested in training flights in conjunction with guidance ground equipment, to examine the correct operation of the guidance chain.

- The two rotor prototypes are used for mechanical tests, wind tunnel tests (Chalet-Meudon), and portal frame tests (Melun). During these tests, experiments are made concerning the flight mechanics of the missile with the rotor unfolded.

Concerning the propulsive unit for cruising flight, an adaptation study is pursued by a specialized company, starting from a fully tested ramjet.

Third Year:

Six missiles are foreseen for carrying out experiments with the unfolding of the rotor and the descent. The guidance loop for the approach and descent under automatic remote control is tested. The climb trajectories of the missile is vertical. The altitude attained is that of the pull-out peak (of the order of 1200 meters).

For these launchings the missile is provided only with a powder rocket of reduced impulse.

If the necessity arises, experiments on the portal frame test stand or in a wind tunnel are taken up again in between the launchings.

Towards the end of the third year, a new series of launchings is taken up. Four missiles of reduced load capacity (provided solely with power rockets of nominal impulse) are launched, in order to test the combination of the two phases "approach" and "descent". In addition, tests of the cruising guidance chain are carried out with aircraft.

Fourth Year:

The flight experiments of the approach and descent phases are completed. Finally, four missiles carrying the complete equipment are launched to test the mission of the normal type (300 km - 50 kg).

NOTE:

The test program which we have just described includes fourteen launchings of missiles. This number appears to us necessary to permit a development of the concept.

It appears to us rather difficult to make assumptions at the present stage of this study, and for so small a number of launchings, about the recovery expectation for the missiles during the course of the tests. We have therefore based our financial estimates on the construction of fourteen missiles.

Two possibilities arise:

- To manufacture all the fourteen missiles and eventually carry out more launchings to simulate the first steps in operations by making use of recovered missiles.

- To reduce the cost of the experimental phase by reaching during the second year a better estimate of the expectation of recovery. In particular, tests of the missile with rotor unfolded in the portal frame test rig should clear up this point.

3 - 2 - FINANCIAL ESTIMATE

In Table I we have given cost figures for each of the headings in the general planning of the development of the missile.

The total amount obtained is:

NF 9,748,000

This price was established, without taxes, based on the economic conditions on the 1st of January, 1962.

Table II gives the sequence of payments (page 25).

NOTE:

In this amount, we have left the following out of account:

- Ground equipment for guidance. In fact, the essential part of this equipment should be supplied by the C.N.E.T.

- Ground equipment, in particular the landing site.

- Equipment concerned with the drones used in training flights.

TABLE 1

Details of Costs

REFERENCE	DESIGNATION	Partial amount in NF	Total amount in NF
E. 1.	Integration Studies		
E. 2.	Studies of Flying Controls and Guidance		
	- Research Staff		1.000.000
E. 3.	Aerodynamic and Stress Analysis		
	- Technical Staff	150.000	
	- Models and Wind tunnel tests	100.000	
			250.000
E. 4.	Development and Manufacture of Components		
E. 4. 1	Missile and Servo-control Components - Power Sources, - Measurements		
	- Design Staff	160.000	
	- Manufacture of equipped airframe	1,920.000	
	- Tooling	250.000	
			2,330.000
E. 4. 2	Rotor		
	- Design Staff	610.000	
	- Construction of test models and wind tunnel supports	264.000	
	- Tests in Eiffel and Sl wind tunnels	200.000	
	- Manufacture of two prototypes	296.000	
	- Manufacture of 14 rotors	1,120.000	
	- Manufacture of 14 driving power units	210.000	
			2,700.000

REFERENCE	DESIGNATION	Partial amount in NF	Total amount in NF
E.4.3	Powder Rocket		
	- Design and bench tests	200.000	
	- Manufacture of 6 rockets with reduced impulse	40.000	
	- Manufacture of 8 rockets with nominal impulse	128.000	
			368.000
E.4.4	Ramjet Units		
	- Modification design and bench tests	370.000	
	- Manufacture of 4 engine groups with fuel supply and automatic controls	192.000	
			562.000
E.4.5	Flying Control and Guidance Equipment		
	- Modification design	220.000	
	- Supply of n assemblies 2 for ground tests and possibly carried flights		
	n - 2 for the missiles		
	Automatic Pilot (16)	1,040.000	
	Remote control receiver	240.000	
	Responder (10)	180.000	
	- Ground tests	250.000	
			1,930.000

REFERENCE	DESIGNATION	Partial amount in NF	Total amount in NF
E.5	Tests		
E.5.1	Wind Tunnel and Portal frame tests of rotor		
	This expenditure was posted under E.4.2		
E.5.2 E.5.3	Carried Flight Launchings		
	- Technical staff travelling expenses	550.000	
	- Ground equipment for launching	58.000	608.000

TABLE 2

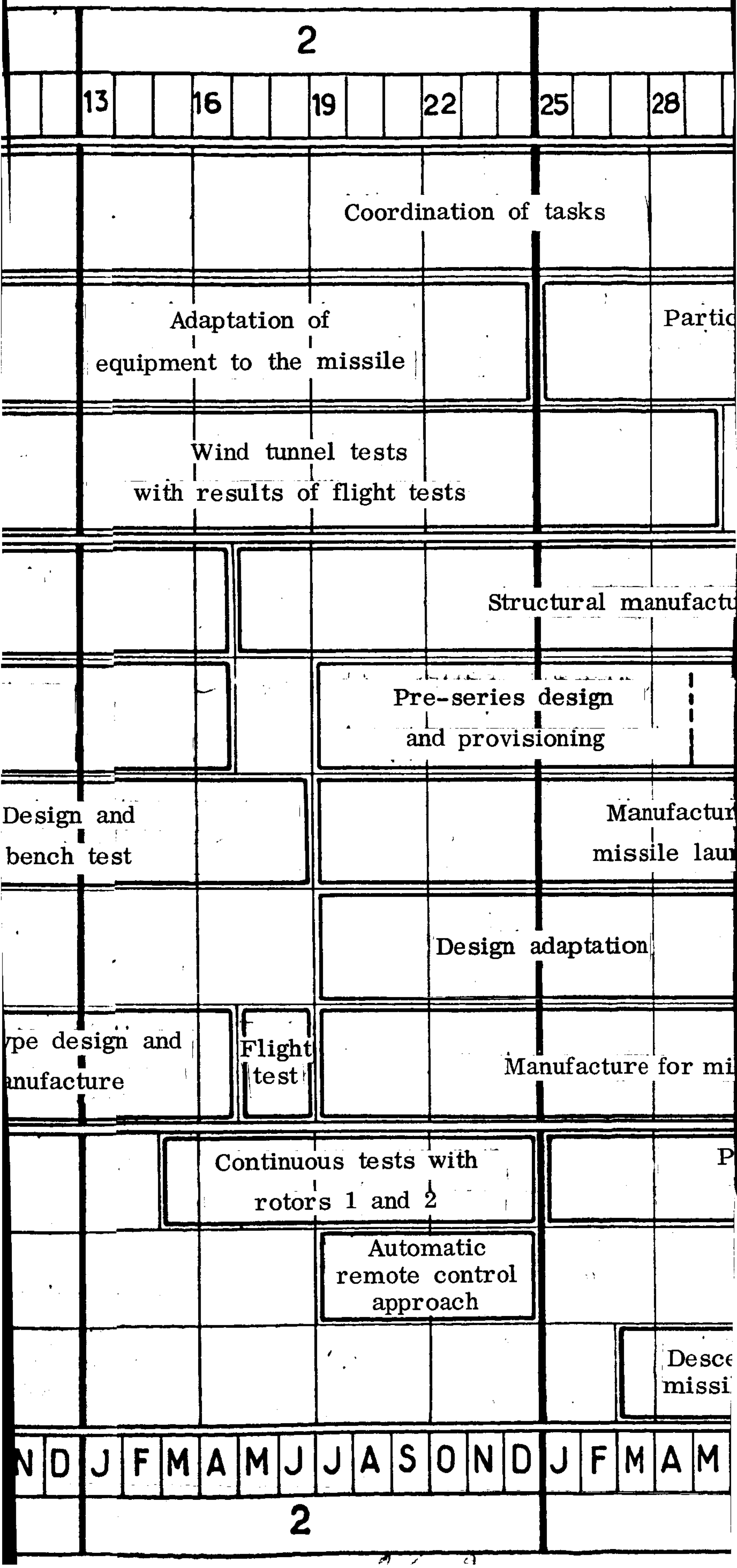
SEQUENCE OF PAYMENTS

REFERENCE	DESIGNATION	Total amount in NF	1st Year	2nd Year	3rd Year	4th Year
E.1	Integration Studies					
E.2	Studies of Flying Controls and Guidance	1,000.000	270.000	330.000	250.000	150.000
E.3	Aerodynamic and Stress Analysis	250.000	150.000	100.000		
E.4.1	Missile and Servo-control components, power sources, measurements	2,330.000	350.000	700.000	800.000	480.000
E.4.2	Rotor	2,700.000	500.000	800.000	1,100.000	300.000
E.4.3	Powder Rocket	368.000	50.000	100.000	168.000	50.000
E.4.4	Ramjet Units	562.000		100.000	400.000	62.000
E.4.5	Flying Control and Guidance Equipment	1,930.000	400.000	500.000	700.000	330.000
E.5	Tests	608.000		108.000	250.000	250.000
	Total	9,748.000	1,670.000	2,730.000	3,718.000	1,630.000

DEVELOPMENT PLA

		YEAR	1
		MONTH	1 4 7 10
E 1	Synthesis (weapon system) studies		Project work
E 2	Flight and Guidance Control Studies		Completion of technical specifications
E 3	Aerodynamic and structural strength Studies		Calculations cross-check
E 4 Design and manufacture of components	1 - Missile and accessories		Structural design and outfitting
	2 - Rotor		Design and manufacture of 2 test prototypes
	3 - Solid - propellant powerplant		
	4 - Ramjet		
	5 - Flight Equipment Guidance Equipment		Proto
E 5 Flight tests	1 - Gantry and wind tunnel tests		
	2 - Captive flight tests		
	3 - Missile launchings		
		MONTH	J F M A M J J A S
		YEAR	1

NING OF THE POSTAL MISSILE



3 | 4

31 | 34 | 37 | 40 | 43 | 46

ipation in tests, captive flights, launchings—.
Exploitation of results

re for launching

Production rotor for missile
launchings (1 per month)

e for
ıchings

Manufacture for
missile launchings

ssile launchings

ossible tests depending
on launch results

Cruise
navigation

nt
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Approach
missiles
4

Complete
missiles
4

J J A S O N D J F M A M J J A S O N D

3 | 4

26-2-7

CHAPTER 4

SPECIFICATION OF THE MISSILE

4 - 1 - AERODYNAMIC CONFIGURATION

The missile configuration is that of a tail-first layout with a cruciform arrangement of the wing system. The two ramjet units are arranged in the yawing plane. Pages 41 and 42 show two general arrangement drawings, the first shows the missile at the start, the second on landing with rotor unfolded.

Fuselage

The fuselage is circular. The front part is an ogive with an aspect ratio of 4. The diameter of the cylindrical part is 0.64 m.

The total length is 5.70 m.

Wings

The wings are of trapezoidal shape, jointed at their root chords two wings of a pair have the following dimensions:

Span	2.56 m
Surface area	2.5 m^2
Root chord	1.46 m
Sweep-back of leading edge	49°

The section profile is lenticular and has a 4% thickness ratio.

The extremities of the wing pair are movable and constitute the control surfaces in roll.

The surface area of the two control surfaces is 0.12 m^2.

Tail First Empennage

The front tail surfaces of trapezoidal shape perform the control of the missile in pitch and yaw.

Surface area of the two front tail surfaces for pitching	0.321 m^2
Surface area of the two front surfaces for yawing	0.212 m^2

This difference of surface areas is due to the disymmetry between the lift values in pitch and yaw. The aerodynamic center in yaw is, in fact, in a more advanced position than the aerodynamic center in pitch.

In order to preserve a sufficient static stability margin in yaw we were obliged to reduce the surface area of the front tail surfaces.

The following curves are shown:

Page 43 The variation of C_x minimum

Page 44 The variation of $\frac{dCz}{di}$ (in pitch) and $\frac{dCy}{dj}$ (in yaw).

Page 45 The variation of $\frac{dCm}{d\beta_t}$ (in pitch) and of $\frac{dCn}{d\beta_1}$ (in yaw) as a function of the Mach number.

As we shall see in Chapter 5, the aerodynamic properties of the missile in a tail wind are very important for the analysis of the dynamic behavior of the missile with the rotor unfolded. In page 46 we show two polar curves of the missile in the "roto-wind" conditions and in page 47 the position of its aerodynamic center in a tail wind.

In addition, in the course of the descent with rotor unfolded, the speed is insufficient to permit control of the missile in roll by the control surfaces at the extremities of the wings. Since, in addition, (this point will be discussed in the chapter on guidance) it is profitable to stabilize the attitude of the missile in roll during the descent, it is appropriate to install a special device for this phase. A variety of means can be envisaged, such as a motor element arranged between the rotating part and the missile, and a micro-nozzle ejecting a compressed gas. A final selection will be made when the friction torque of the rotor hub will be determined more precisely. However, within the framework of the present document, we have provided the missile with a stabilizing device operating with compressed air which acts through two nozzles placed at the ends of the wings.

4 - 2 - PROPULSION

4 - 2.1 - Powder Booster Rocket Stage

The propulsive unit has the following data:

Powder	Isolane
Specific Impulse	$I_{sp} = 238$ sec
Weight of powder	360 kg
Empty weight	80 kg
Impulse	$840 \cdot 10^3$ N. sec
Outside diameter	640 mm
Overall length	1930 mm

Final analysis of a more advanced type will make it possible to optimize the duration of propulsion which was selected to be 10 secs for the present project study. It should be noted that, after 4 secs, in other words, at $M = 0.8$, the ramjet units intervene during the boosting phase to an extent which is no longer negligible. The lower strata of the atmosphere permit the ramjet unit to develop a large thrust in spite of the substantial reduction of the thrust coefficient at the operating Mach number.

4 - 2.2 - Ramjet Cruising Stage

Description

Each Ramjet unit has the shape of a circular elongated barrel. The front portion constitutes the diffuser surrounding a central body with a conical nose having an apex semi-angle of about 30°. The cylinder diameter is 0.400 m. The rear portion contains the convergent and divergent nozzle. The total length of the propulsion unit is 2.500 m. The empty weight of each propulsion unit is 45 kg, without counting the fuel supply and control devices.

The required injection pressure will be supplied by a turbine fed by outside or inside air.

The design of the ramjet unit will be entrusted to a specialist company. The performances required of this engine are within the framework of French built engines and should not, therefore, require new development.

Performances

Curves of the thermodynamic thrust coefficient C_T (see page 48), referred to the cross-section of the combustion chamber correspond broadly to a combustion temperature of 2000°K. This combustion temperature is achieved and with a mean fuel enrichment of the mixture of about M = 0.90 and a combustion efficiency of η_b = 0.85, approximately.

The thrust coefficients adopted are very nearly those of the Nord-Aviation ramjet unit which powers the Matra R.431 missile.

The specific fuel consumption applicable to a Mach number adjusted to altitudes above Z = 11,000 m is of the order of $0.80 \cdot 10^{-4}$ kg/N sec.

For Mach numbers below or above the adjusted Mach number, the specific fuel consumption increases to:

$$C_{sp} = 2.00 \cdot 10^{-4} \text{ kg/N.s for } M \simeq 1$$

This shows the attraction of utilizing the ramjet unit during the acceleration phase of the missile (the specific fuel consumption of the powder rocket is $C_{sp} = 4.11 \cdot 10^{-4}$ kg/N. sec).

4 - 3 - ROTATING WING

4 - 3.1 - Rotor Design

Page 49 shows a drawing of the general arrangement of the rotor.

The rotating wing includes two blades, each consisting of three retractable elements of which the two extreme elements are telescopic.

The outside element consists of a hollow casing, stiffened by a grid consisting of radial stringers and transverse stiffeners in stainless steel which provide the stiffness of this component in torsion and flexure. The centrifugal force is resisted partially by the duralumin skin and another part of the centrifugal force is transmitted by a central tie beam which rests against the Congeron of the intermediate element.

The intermediate element is filled. Its longeron is made of "Durisol," the skin of stainless steel, a filling in "Frigolite" ensures the stiffness of the rear portion of this component.

Concerning the central components, we may note that, in order to eliminate the critical condition of ground resonance, the blades are stiffened in their plane of rotation by means of two preloaded stays. These stays serve at the same time for the transmission of the engine torque. The largest part of the centrifugal force (80%) is taken up by the central tie rods which permit the displacement of the blade about its torsion axis (blade pitch variation).

The blades are attached to the rocking hub which is gimbal mounted on the upper part of the torque tube. The lower part of this tube carries two hinged arms which have at each of their extremities a rocket engine.

The blades are controlled by a swashplate which permits a collective and cyclic variation of the blade pitch. For this purpose, a link-work connects the rotating part of the swashplate to each blade. The axial movement is transmitted by the torsion of an arm pivoted on the casing which forms the root of the blade. The non-rotating part of the swashplate is controlled by three jacks which form part of the automatic pilot.

The particular arrangement of the control jacks is such that the action of one of the three jacks corresponds to pitching control. The differential movement of the two other jacks causes a yawing response of the rotor. At the same time, the simultaneous displacement of all three jacks gives rise to a variation of the collective pitch of the rotor.

4 - 3.2 - Schedule of Weights

The weights schedule given below includes solely the part concerning the rotor, its controls, and its power transmission.

Outside blades	2 x 3.800		
Intermediate blades	2 x 4.500		
Blade arms	2 x 4.600	2 x 15.900	
Drag stays and hinges	2 x 1.500		
Torsion tubes	2 x 1.500		
Balancing beam	2.900		
Hub	6.500	18.400	
Swashplate	7.000		
Control rods	2.000		
Total, comprising the rotor proper			50.200 kg
Power transmission	3.500		
Locking system and the opening and folding mechanism of the rotor	5.500		13.500
Control jacks	3 x 1.500		
			63.700 kg
Sundries 10%			6.300
			70.000

The total weight given above contains a reserve of 10%.

4 - 3.3 - General Characteristics

Total weight for recovery (including rotor)	600 kg
Maximum load factor	2.5
Driving torque	1400 N.m
Number of blades	2
Rotor diameter	6 m
Chord (0012 profile)	0.3 m
Tip speed (F = 120,000N)	200 m/s
Maximum tilt of the rotor (cyclic swashplate)	$\pm$ 20°
Collective pitch variation	from -18 to +16°
Twisting torque of one blade	250 N.m

4 - 3.4 - Rotor characteristics in missile application

Hovering Power Required

Its variation as a function of the tip speed is shown in p. 50, as well as the driving power available for a torque of 1400 N.m.

Power required in level flight

Its variation as a function of the horizontal forward speed is shown in p. 51, for a tip speed of 200 m/s. This variation shows that for the available driving torque, the maximum level flight speed is about 19 m/s.

Polar speed curves

In the diagram of p. 52, we give the polar curve in autorotative descent, from which the minimum slope of descent of 40.5° and a minimum rate of descent of 19 m/s can be read for the case of an axial flow.

The polar curve for powered descent is also shown and indicates a maximum flight speed of 19 m/s for the case of level flight.

4 - 4 - AIRFRAME AND EQUIPMENT

Page 53 shows a drawing of the equipped airframe.

4 - 4.1 - Fuselage

The fuselage consists of the following components:

Ogive — Made of duralumin sheet, it carries the empennages. It contains all the flying control equipments in front of the frame which supports the empennages. At the rear of this frame, control surface actuating jacks and rotor pitch control jacks are mounted. This ogive is mounted at the end of the central tube.

The central tube is made of steel, has a diameter of 180 mm and it is integral with the central part. It serves as a support for the rotor hub. We have utilized its internal volume for placing the compressed air reserve required for feeding the nozzles which control the rolling attitude of the machine during the descent with rotor unfolded.

The central part — It is made of duralumin sheet. It contains in its front a sealed compartment which constitutes the kerosene tank. The central tube is supported on the frames forming the end walls of this tank. The powder rocket is placed at the rear of the tank. This rocket is supported by two solid frames which also carry the attachments of the wings and the ramjets.

The rear drum — Surrounding the elongated tube and the powder rocket nozzle, the drum constitutes the missile base. The freight is placed inside the drum (available volume of about 75 liters). The drum is easily detachable to permit the replacement, after each flight, of the powder rocket.

The rotating part — The rotor designed by the Giravions Dorand Company rotates on the central tube of the fuselage. In its folded position it is enclosed in a cowling which forms the continuation of the fuselage shape and which rotates with the rotor. This cowling carries two flat arms which seal the blade nests. In the open position, they transmit the engine torque. A liquid "monergol" engine such as a Napier hydrogen peroxide rocket developed for use in helicopters, will be placed at the extremity of each of the two flat arms.

In the experimental stage, this arrangement was preferred to the placing of the engines at the end of the telescopic blade elements. Such a solution would be more efficient from the point of view of power utilization but would require a more complex fuel supply system.

4 - 4.2 - Wings

The wings are of classical design in duralumin sheet. They carry a landing leg each, containing a shock absorber designed to absorb the energy corresponding to a vertical rate of descent of 3 m/s. Two wings situated in the same plane, have ailerons at their extremity for control in rolling. The two other wings are provided with micro-nozzles supplied with compressed air. The nozzles are at right angles to the plane of the wings. The nozzles overcome the friction torque of the rotor and control the orientation of the machine during its descent as a helicopter.

4 - 4.3 - Ramjet units

The ramjets are arranged symmetrically on each side of the fuselage. The central body in one of the two ramjets contains the pumps and the control equipment required for their fuel supply. The central body of the other ramjet contains the power group which supplies the hydraulic servomotors. The ramjets are attached to the fuselage by profiled beams. One of these contains the accumulator batteries and the other the guidance equipment.

All the components are accessible by easily removable doors which permit an easy performance of inspection and exchange operations.

4 - 5 - SCHEDULE OF WEIGHTS AND CENTERS OF GRAVITY

The components which make up the weight schedule have been grouped in the following assemblies:

Airframe

- Fuselage	93 kg
- Equipped rotor	70 kg
- Wings	100 kg
- Ramjets	90 kg
- Rocket, empty	80 kg

Equipment

- Flying control equipment (sensing elements, control assembly, hydraulic power assembly)	35 kg
- Guidance equipment (remote control receiver, responder)	12 kg
- Batteries	15 kg
- Ramjet pump and control unit	15 kg
- Wiring, hydraulic circuit piping	19 kg
- Rotor power units	13 kg
- Compressed air stabilizing device	6 kg

Fuels

- Compressed air	12 kg
- Liquid "monergol"	18 kg
- Kerosene	190 kg
- "Isolane" powder	360 kg

Freight

The following total weights result from this schedule.

Starting weight	1170 kg
Weight after the end of the powder combustion	810 kg
Weight at the emergence of the rotor	620 kg
Landing weight	590 kg

The corresponding center of gravity distances are indicated on the general arrangement drawing shown on p. 41.

4 - 6 - PERFORMANCES

Page 54 shows the typical trajectory of the missile.

In page 55 we have traced the speed and altitude variations as a function of the flying time.

The missile arrives vertically above the landing point after a flying time of 580 secs. The descent with rotor unfolded is carried out in 80 secs. The missile carries out the mission of 300 km in 11 min.

In order to judge the effect of an increase in the freight weight conveyed or of an error in the initial weight schedule, we have established the performances of a missile of which the empty weight is larger by 30 kg than the present weight schedule. To carry out the mission of 300 km, such a missile will have a launching weight of 1222 kg which is an increase of 52 kg over the present weight. On the other hand, assuming the same launching weight, the performance will not exceed a distance of 230 km.

The typical trajectory followed by the postal missile can be split up to into four phases.

Climbing Phase

The start proceeds vertically. The missile remains vertical during 15 secs. During the first 10 secs, the missile accelerates under the action of the powder rocket thrust and the ramjet thrust. After the powder rockets have burnt out, the missile attains a speed of 700 m/sec. and an altitude of 300 meters. The maximum acceleration is 9 g.

After the extinction of the powder rocket, the ramjet thrust balances substantially the drag and the weight so that the missile maintains a constant speed.

At an altitude of 7500 meters (t = 16 secs), the missile starts a flight maneuver with a small incidence at the end of which it reaches an altitude of 20, 000 m in horizontal flight, at a Mach number of about 2. The distance traversed in the horizontal plane is X = 15 km.

The fuel consumption of the ramjets during that phase is of the order of 60 kg of kerosene. This is due to the fact that the ramjets operate continuously with a fuel enrichment ratio of 1 in a region of high dynamic head. This is nevertheless economical because as we have seen earlier, weight by weight, the kerosene delivers a larger impulse than the powder.

Other modes of climbing have been envisaged. Thus a slightly inclined take-off followed by a ballistic flight is less profitable in the performances obtained. This is due to the fact that on such a trajectory, the missile moves longer at a high dynamic head. The gain in the horizontal distance traversed is not sufficient to counterbalance the increase in fuel consumption which results. Moreover, so far as the vertical take-off is concerned, the missile, making use of a large wing surface, is not subject to prohibitive induced drags during the flight maneuver of entering horizontal flight.

Cruising Phase

The missile carries out this phase in steady state flight at a constant speed and altitude. The chosen values, $M = 2$ and $Z = 20,000$ m are of the order of magnitude which should be specified when the final choice of the ramjet is made.

During that phase, the missile traverses about 260 km and its kerosene consumption amounts to 120 kg. Its weight at the end of the cruising flight is 630 kg.

Approach Phase

At 25 km distance from its arrival point, the missile carries out a diving maneuver which finishes up in a descent along a straight line inclined at 45°. In this way, the missile reaches a low altitude (750 m) and low speed (250 m/sec.).

Starting from this point, the missile carries out a nose lift maneuver after which it is found in a vertical position at an altitude of 1200 m and a speed of the order of 50 m/sec. vertically above the arrival point.

This speed is small but, since the altitude is low, the corresponding dynamic head nevertheless is sufficient to ensure the control of the missile.

At this point the control signal is given to unfold the rotor.

Descent Phase with Unfolded Rotor

We mention this phase by way of a reminder since it will be the subject of a detailed description in the following section under 5 - 4.

Let us note that this phase can be split up into three parts:

- Braking of the climbing speed and establishment of the speed of descent in autorotation (12 secs)

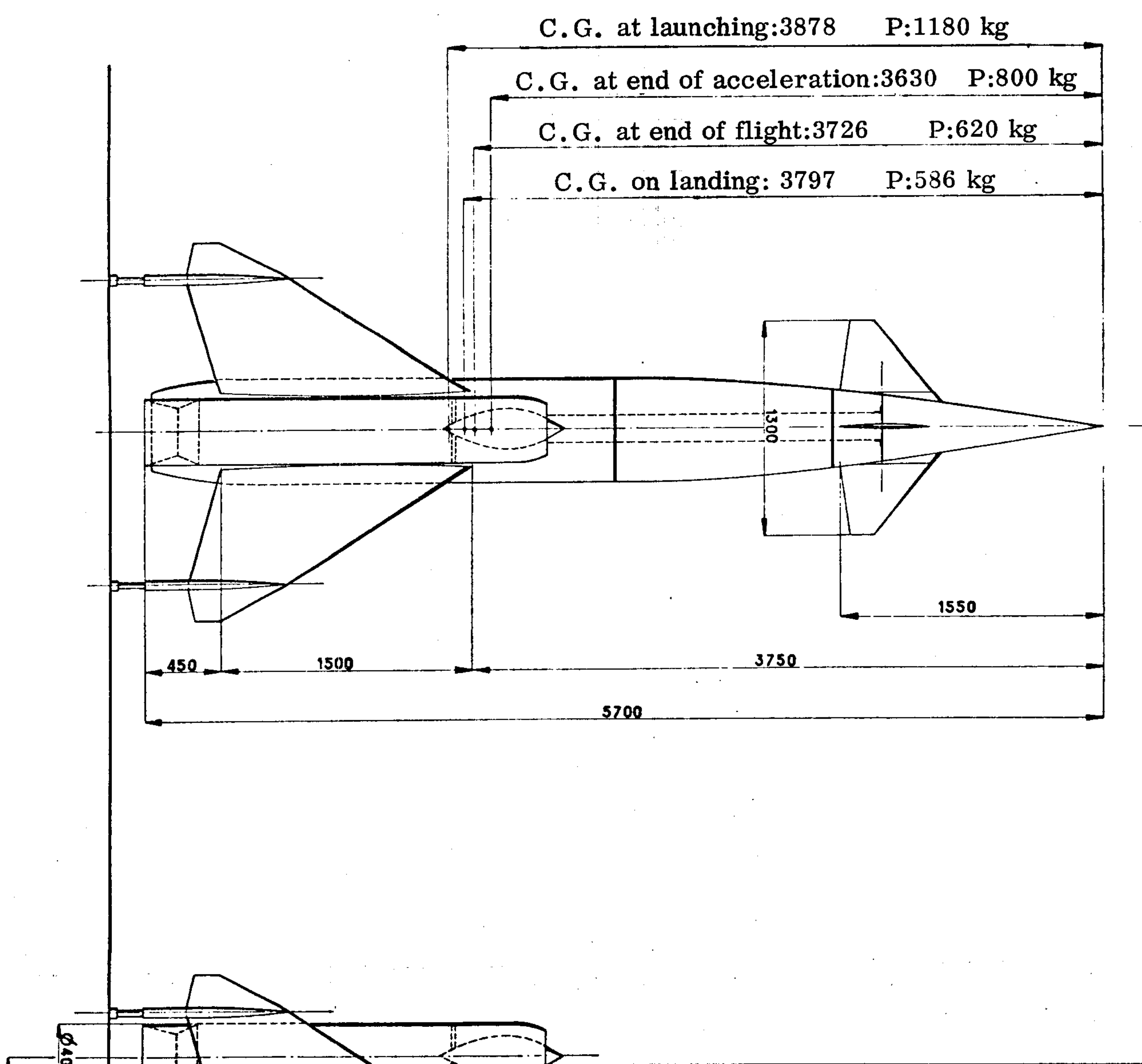
- Descent in autorotation (65 secs) at the rate of 20 m/sec before touchdown.

Note - Ultimate developments

If a larger stage length is required, the present concept can be further developed without major difficulties.

Thus, since the cruising fuel consumption is 0.42 kg/km, an increase of the stage length by 200 km can be obtained by carrying about 100 kg of kerosene more and by increasing the impulse of the powder rocket correspondingly.

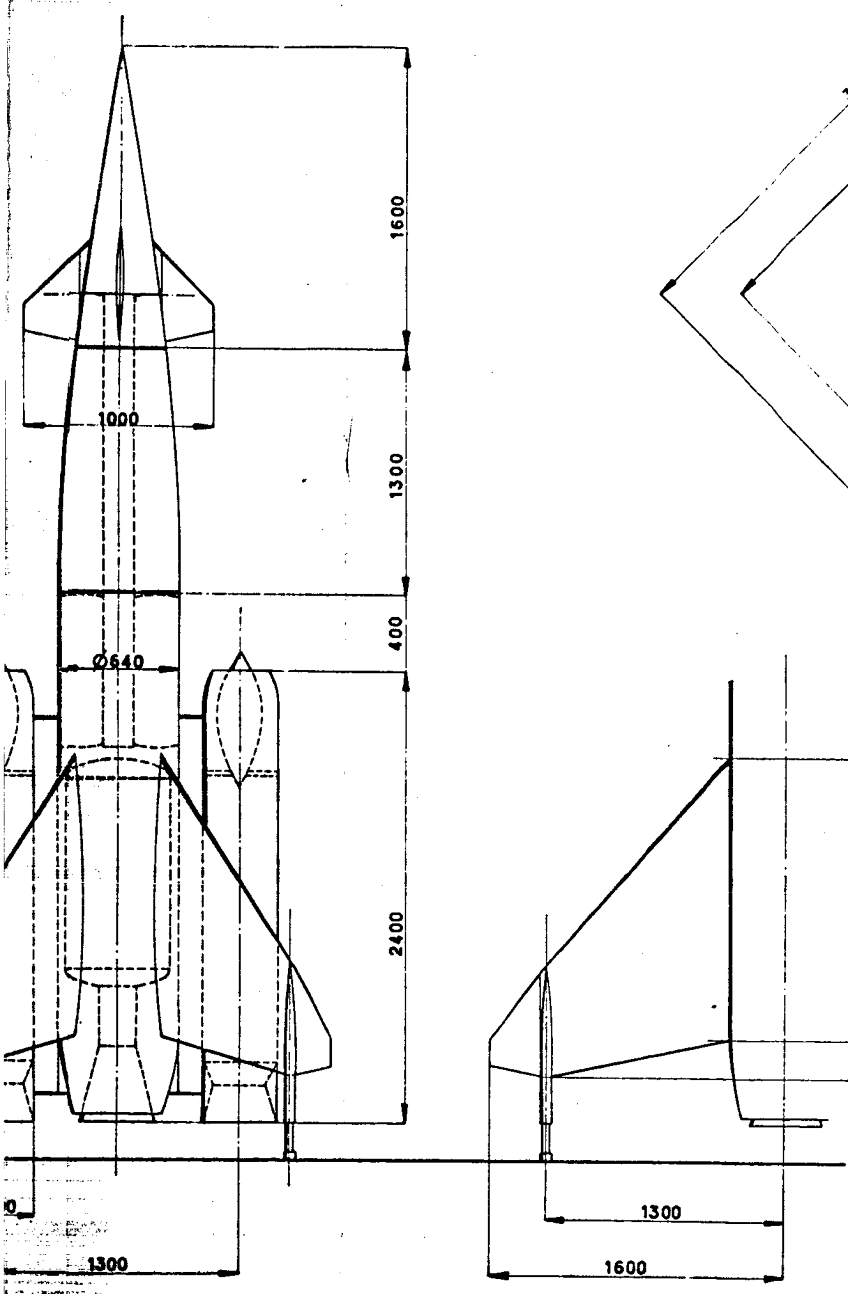
Moreover, raising the level flight speed (to M = 3, for example) permits a reduction of the cruising fuel consumption. The powder and kerosene consumptions during the climb phase will increase (acceleration until M = 3). However, since the proportion of the cruising phase increases within the complete mission, the net balance will certainly be a gain.



C.G. at launching:3878 P:1180 kg
C.G. at end of acceleration:3630 P:800 kg
C.G. at end of flight:3726 P:620 kg
C.G. on landing: 3797 P:586 kg
1300
1550
450
1500
3750
5700

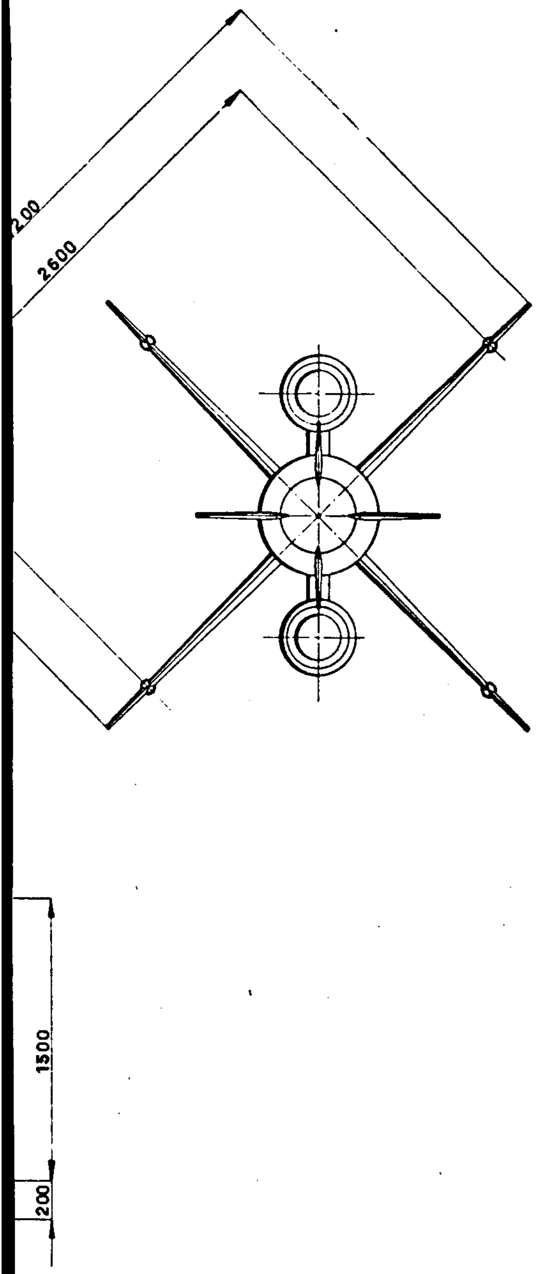
41-1

ENGINS MA

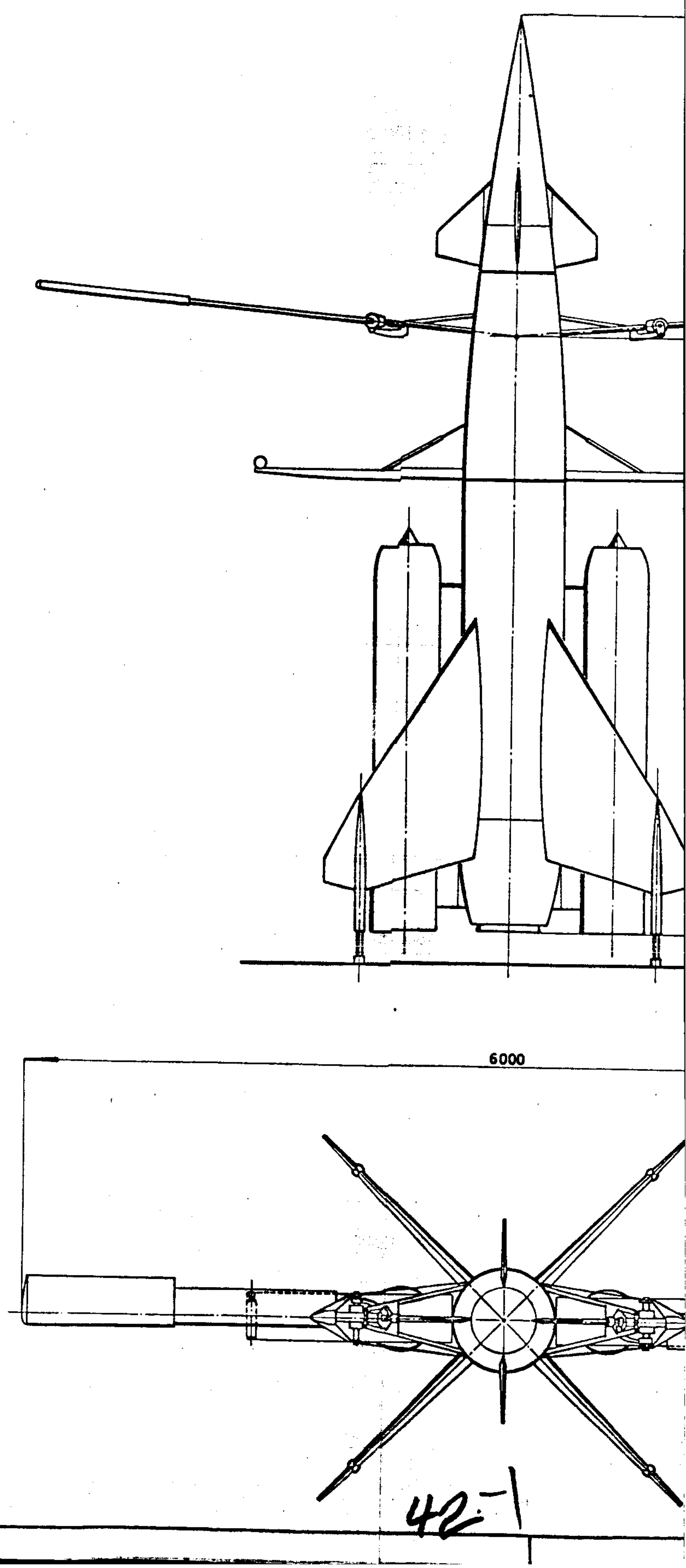


41-2

ATRA	Postal Missile	- 41 -
	Missile on launching	

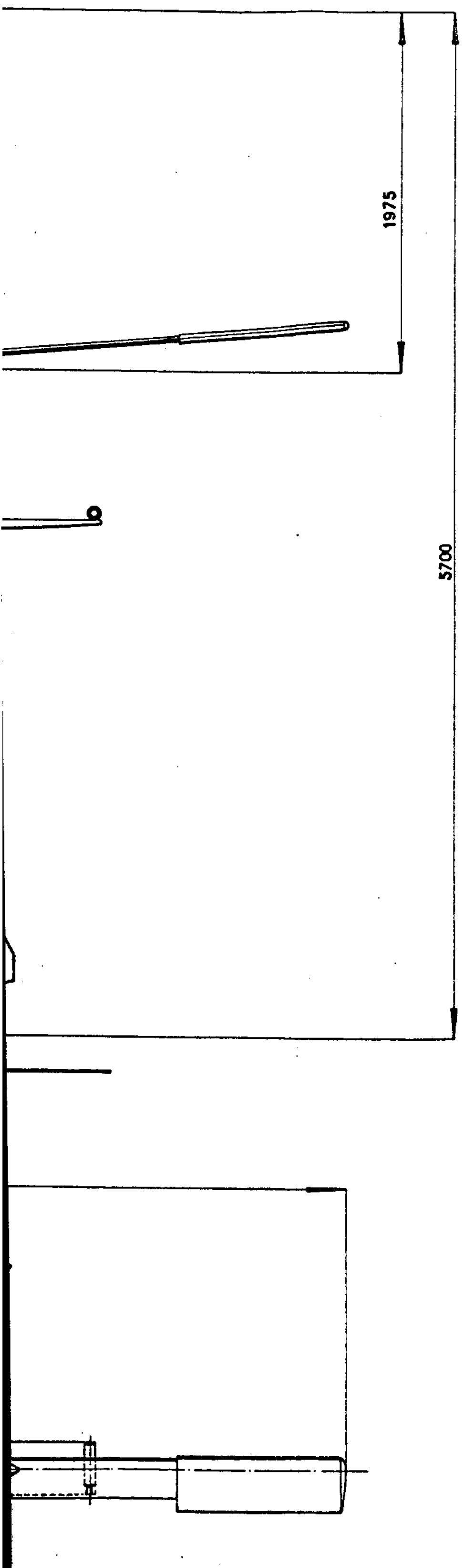


41-2 3

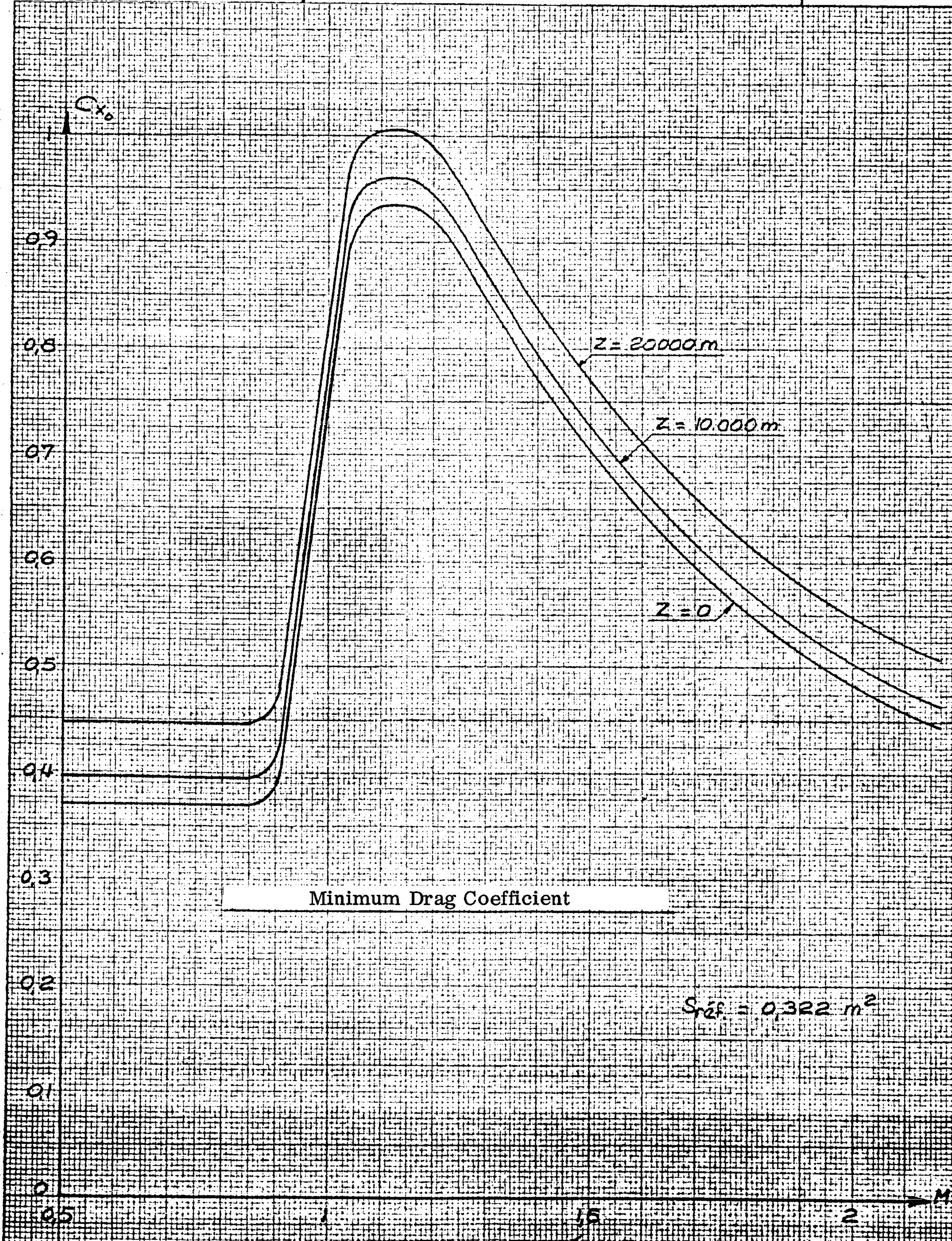


42-1

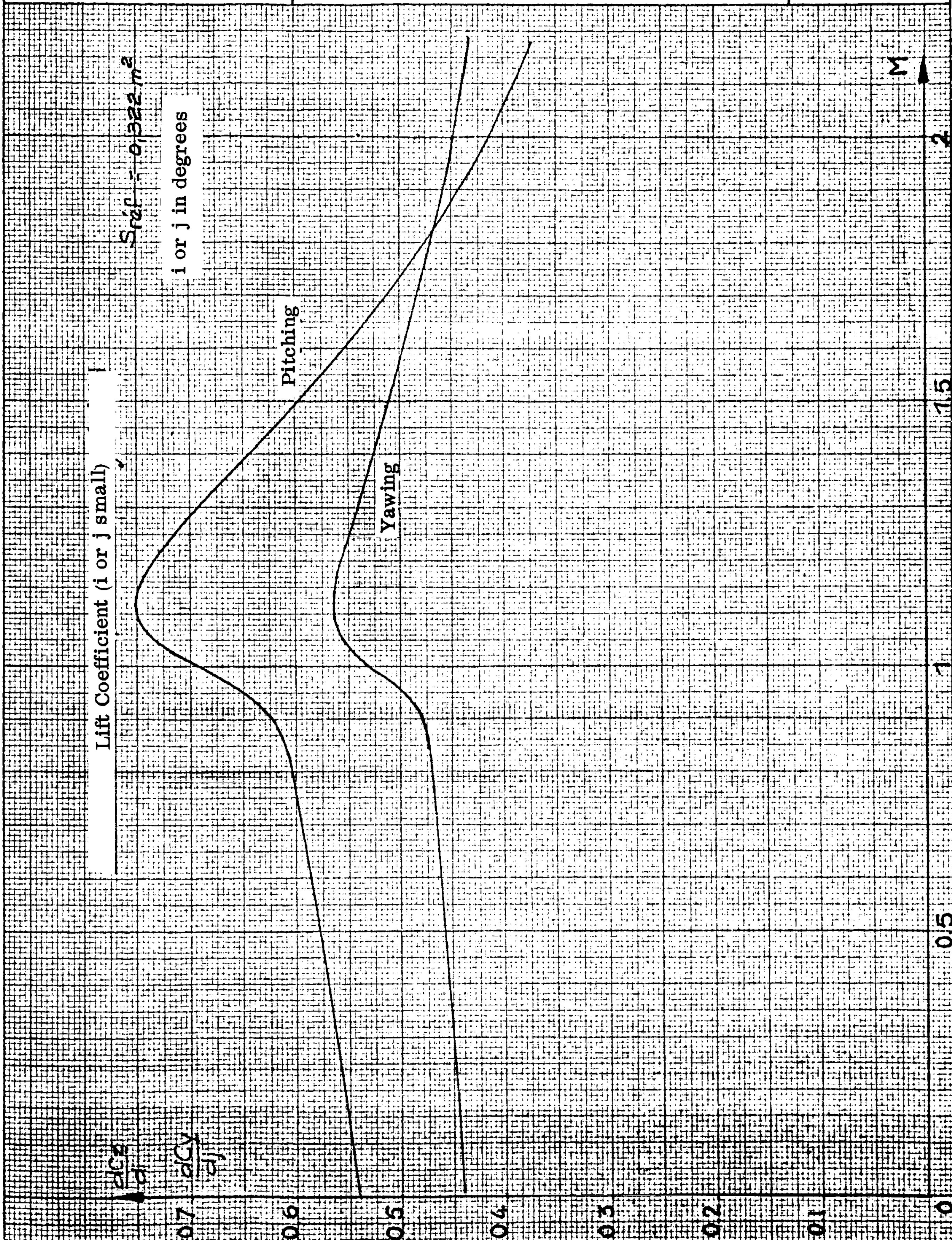
ENGINS MATRA	Postal Missile	-42-
	Missile on Landing	

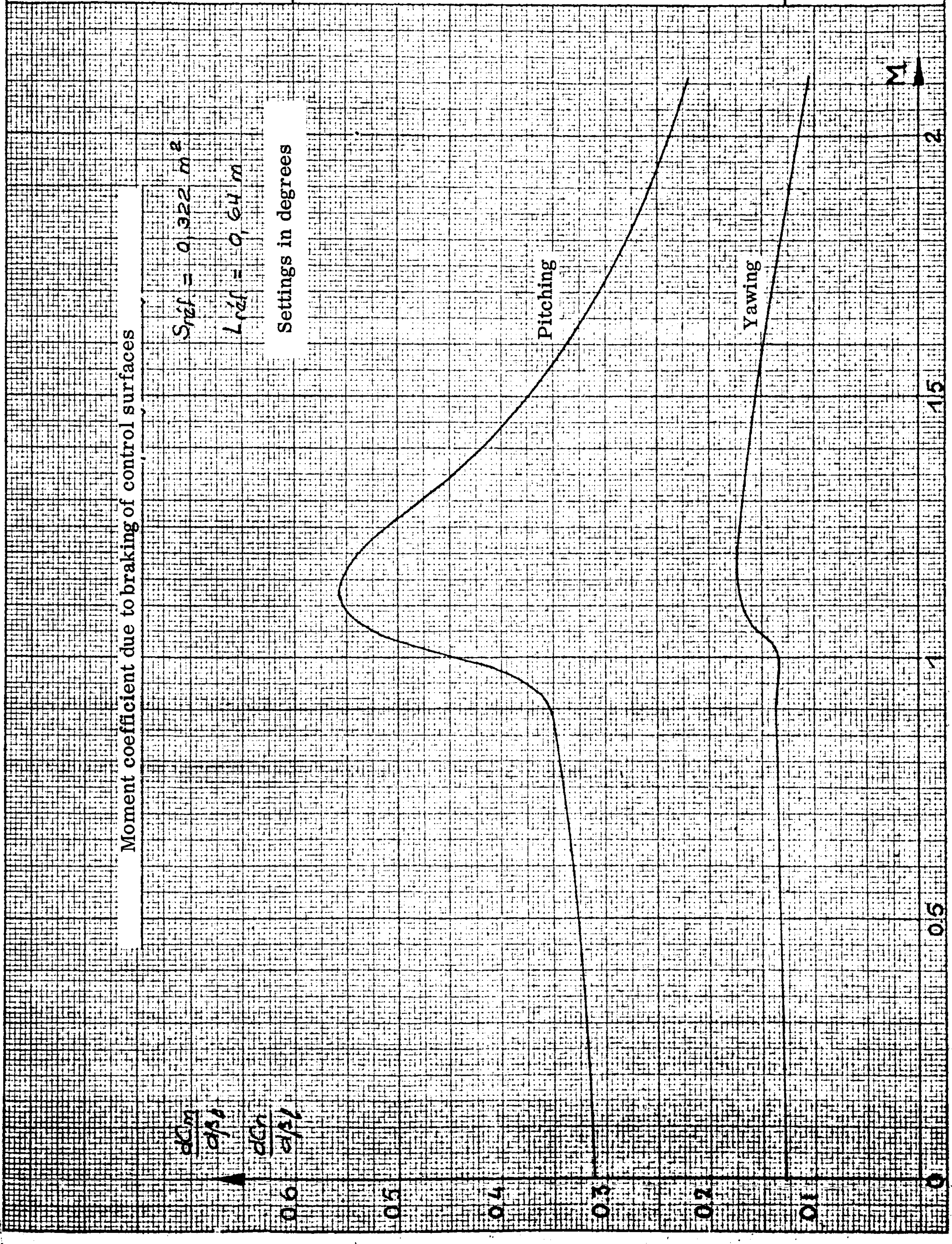


42-2

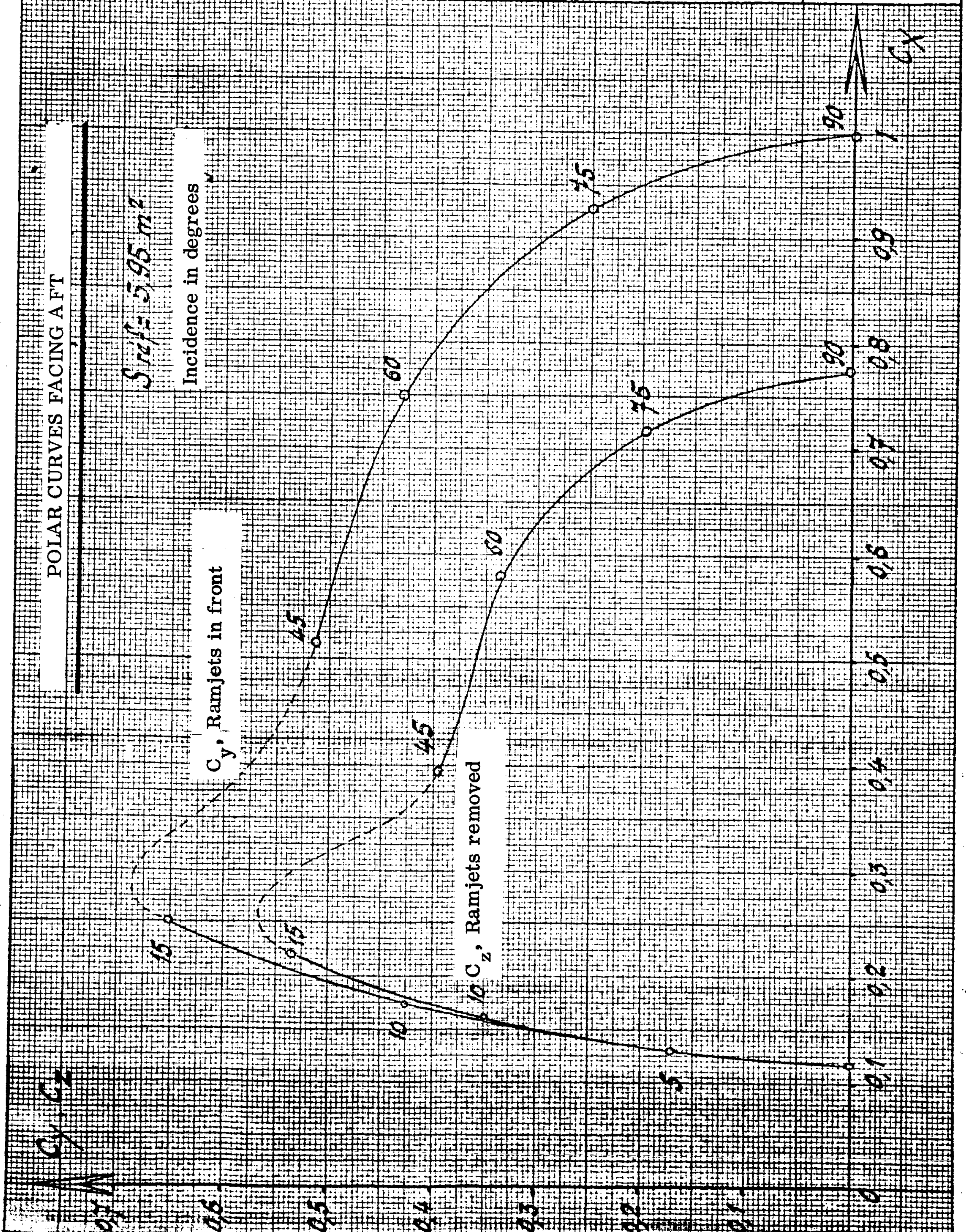


Minimum Drag Coefficient

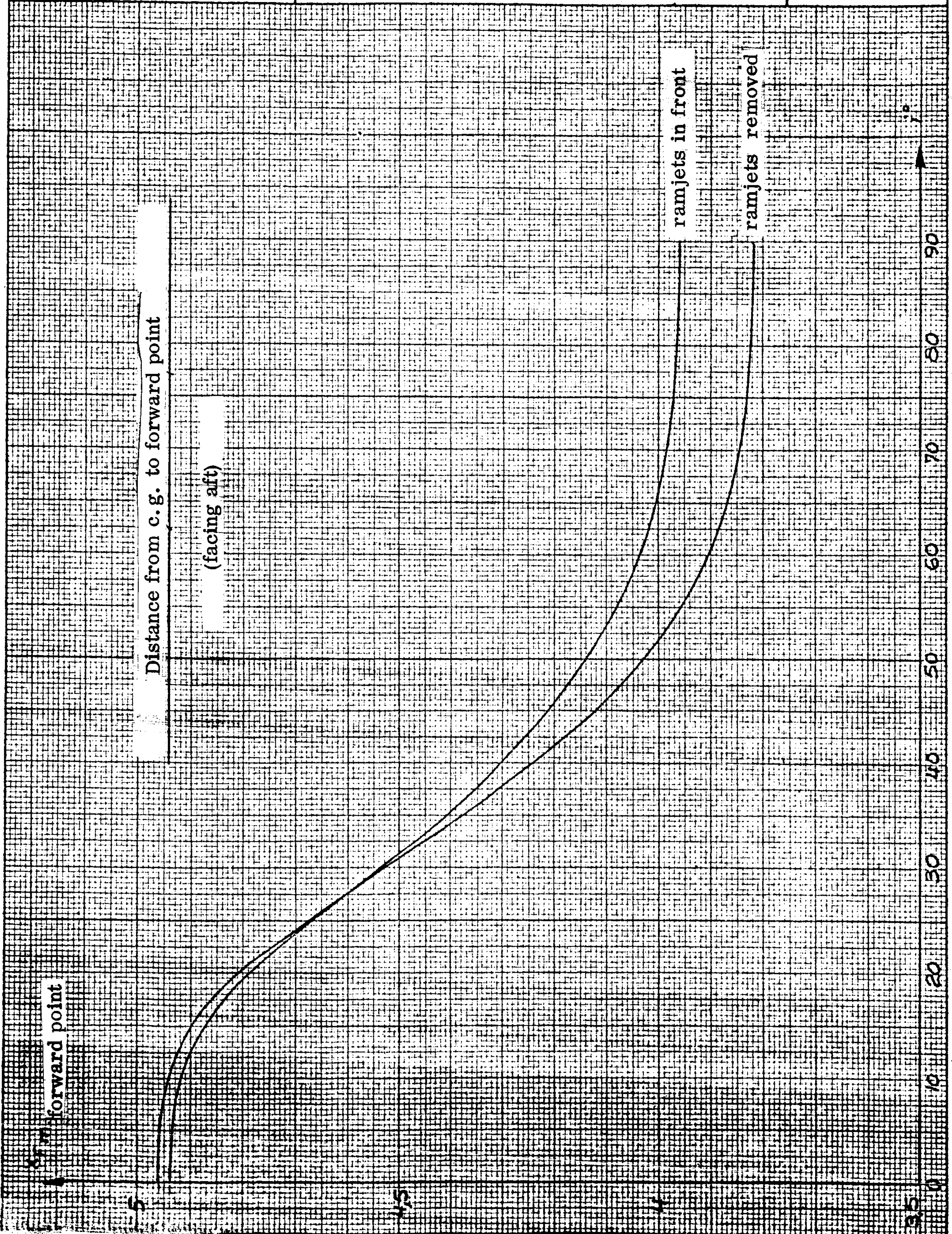

Lift Coefficient (i or j small)
i or j in degrees
Sref = 0,322 m²
dCz/di
dCy/dj
Pitching
Yawing
M
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.5
1
1.5
2



Moment coefficient due to braking of control surfaces
$S_{réf} = 0,322 \ m^2$
$L_{réf} = 0,64 \ m$
Settings in degrees
Pitching
Yawing
$\frac{dCm}{d\beta_1}$
$\frac{dCn}{d\beta_1}$
0.6
0.5
0.4
0.3
0.2
0.1
0
0.5
1
1.5
2
M

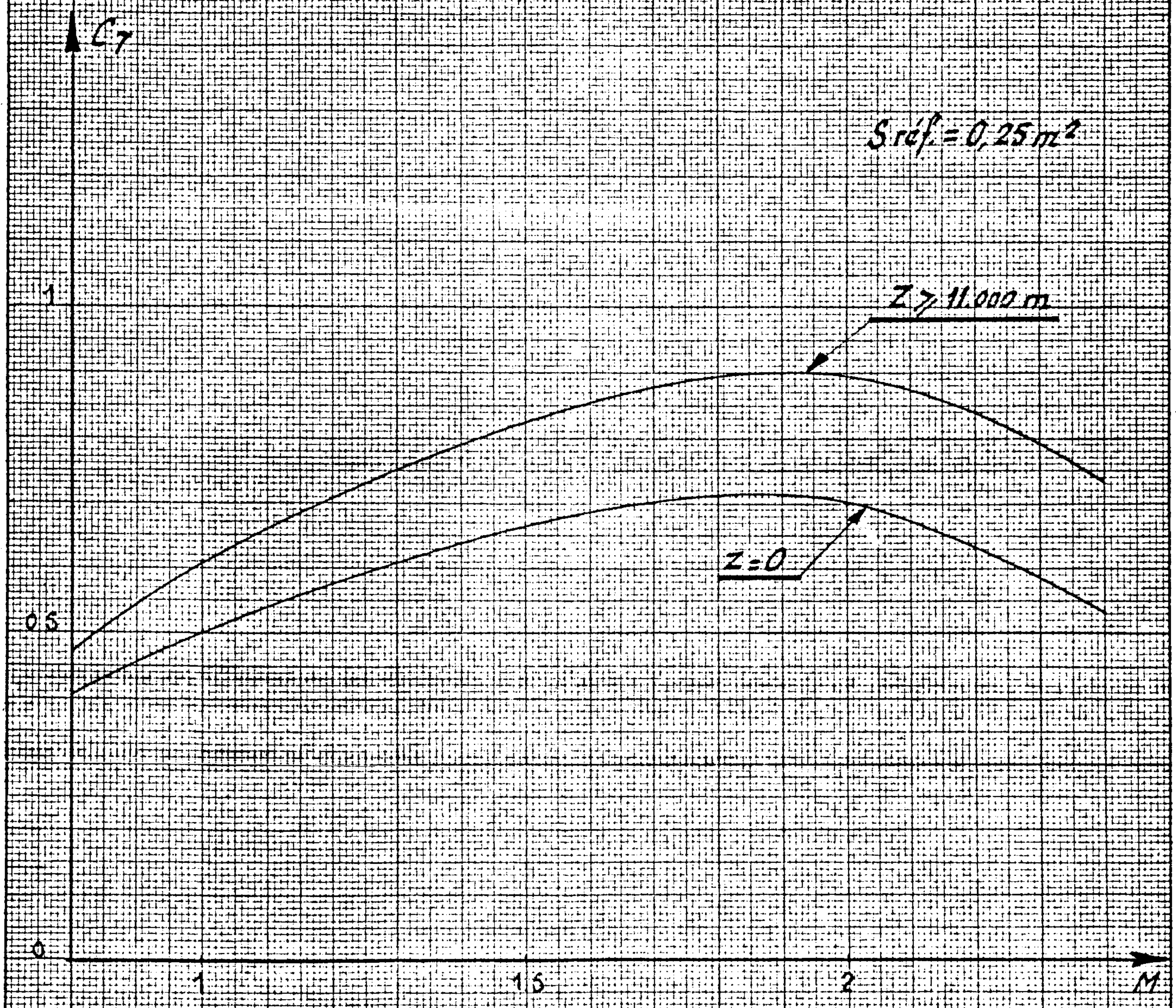


POLAR CURVES FACING AFT
S ref = 5.95 m²
Incidence in degrees
Cy, Ramjets in front
Cz, Ramjets removed
Cx
Cy Cz

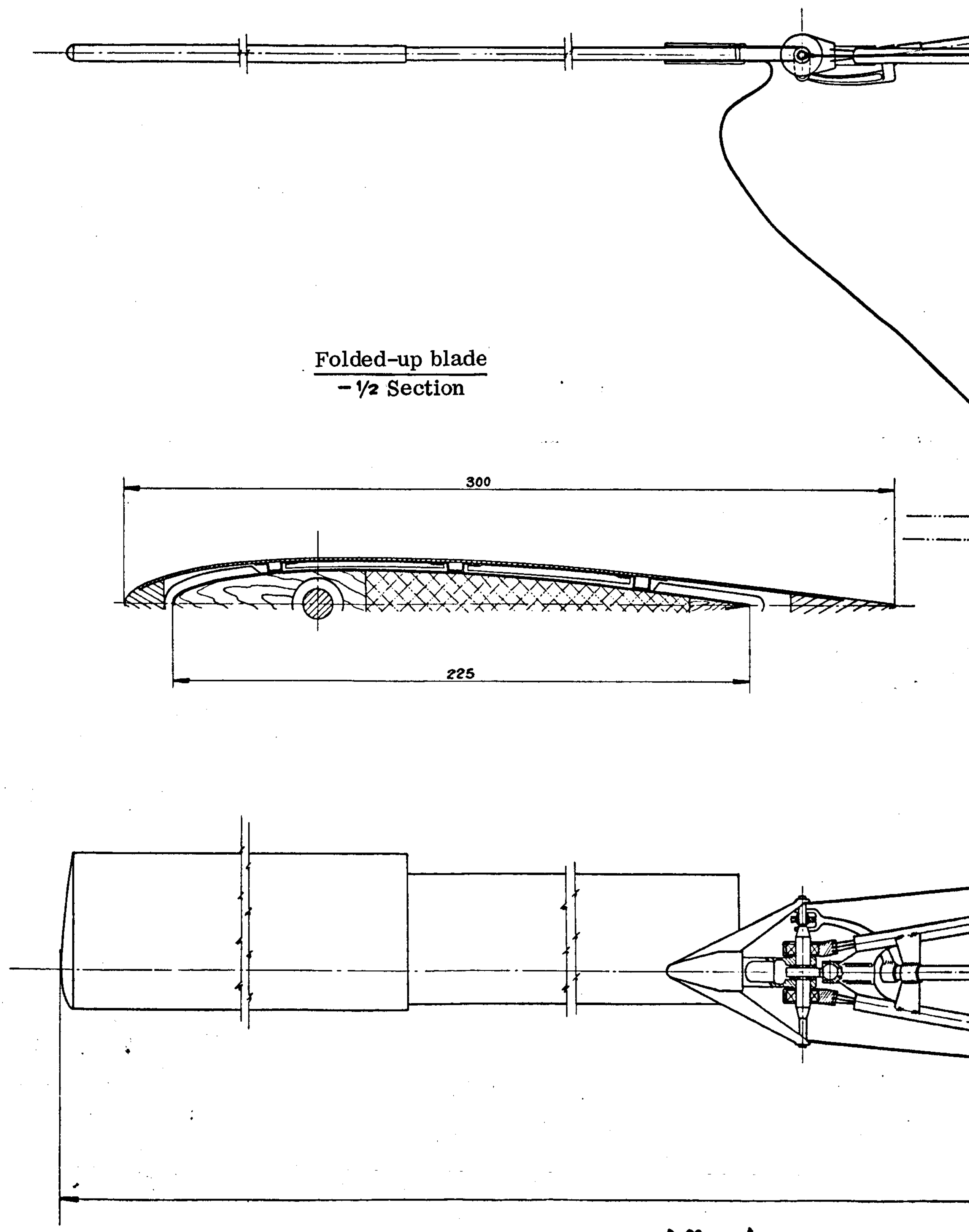

ramjets in front
ramjets removed
Distance from c.g. to forward point
(facing aft)
forward point
0
10
20
30
40
50
60
70
80
90
5
4,5
4
3,5

Thrust Coefficient of Ramjets

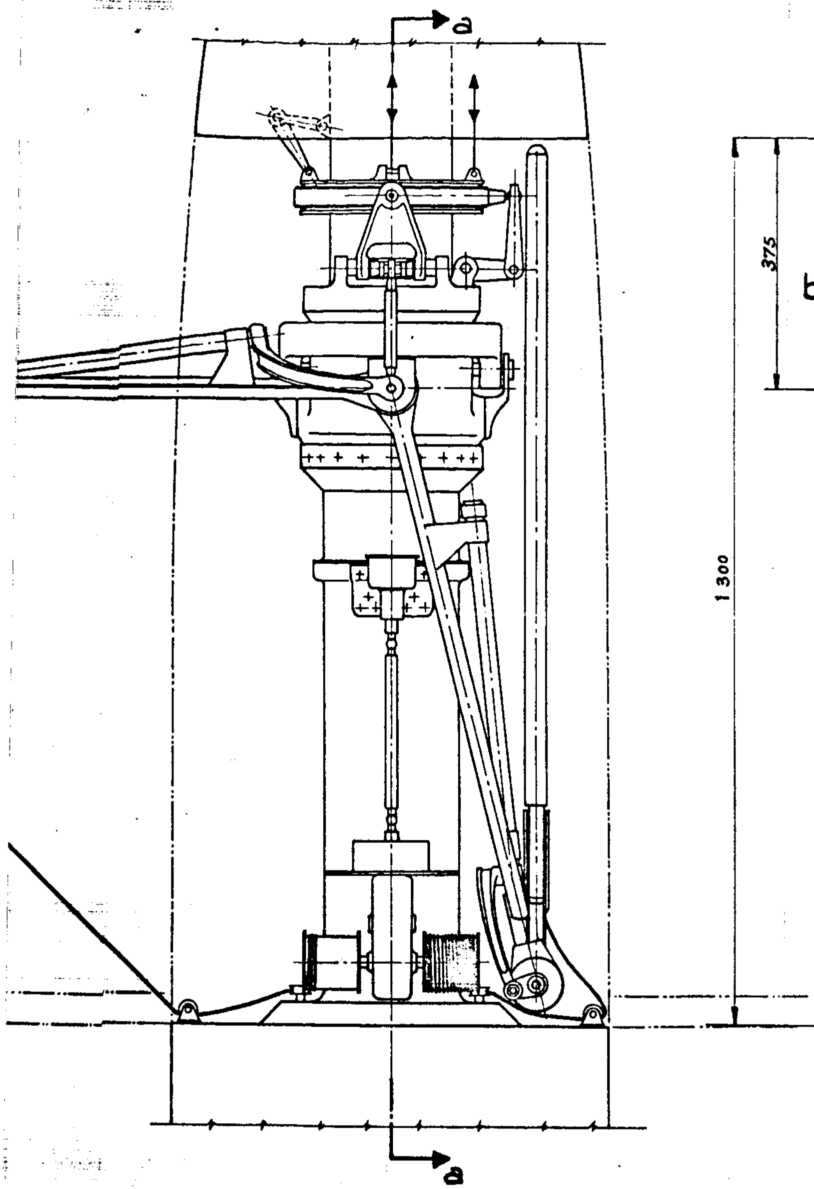
(Stochiometric)

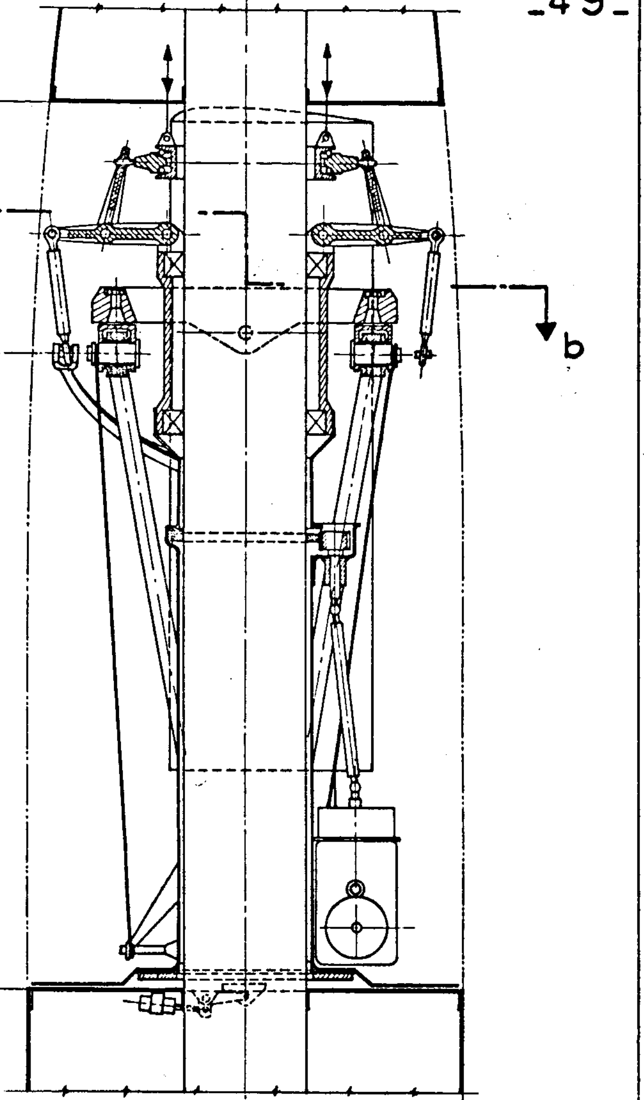


laisser la marge de ce coté



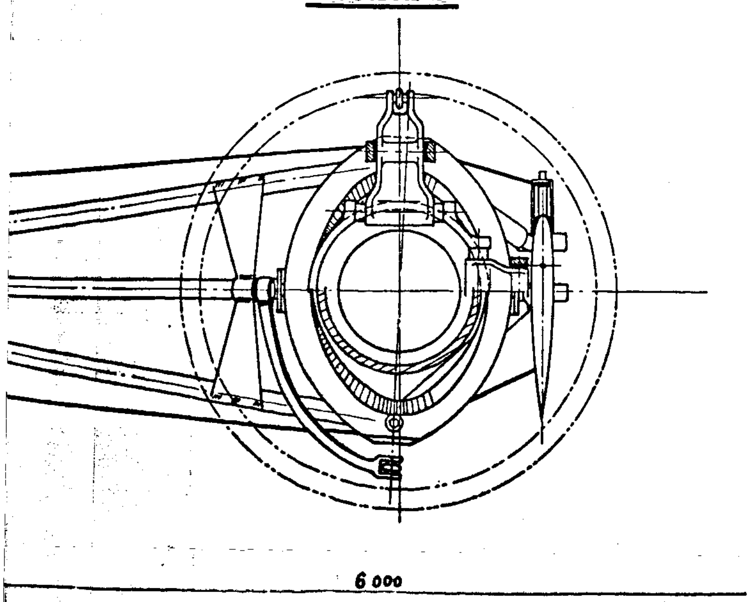
49-1





Section a

Section b



Landing rotor for MATRA Missile proj. Survey

Ref.	Nº # piece	Designation	Dimensions	mater.	Plan #

DORAND GYROPLANES

Scales ___ # of pieces per machine

Weight ___

1961 Name Date Allotment

Desg. Ch. [illegible] 21.11

Cert.

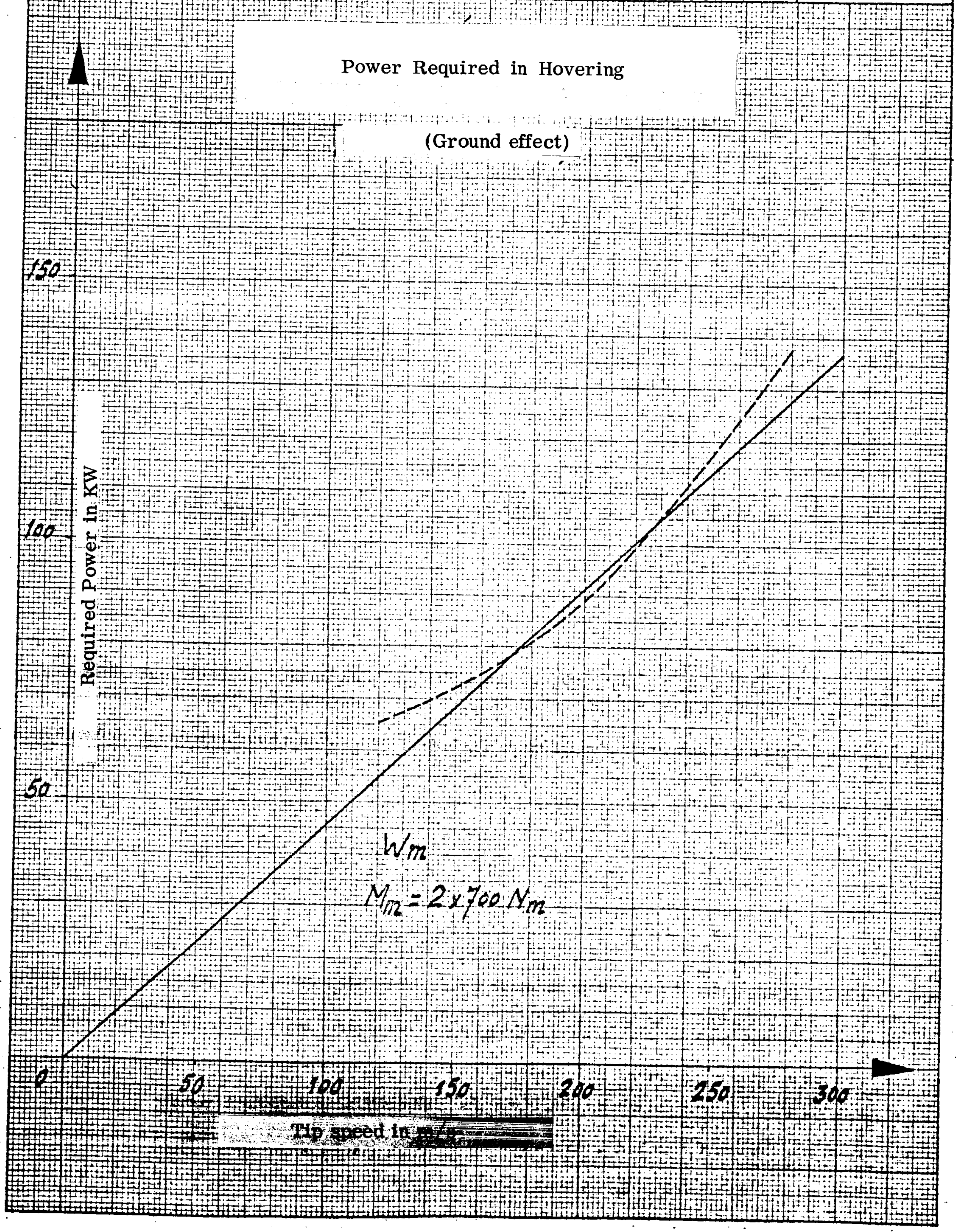
Landing Rotor for MATRA Missile project Survey

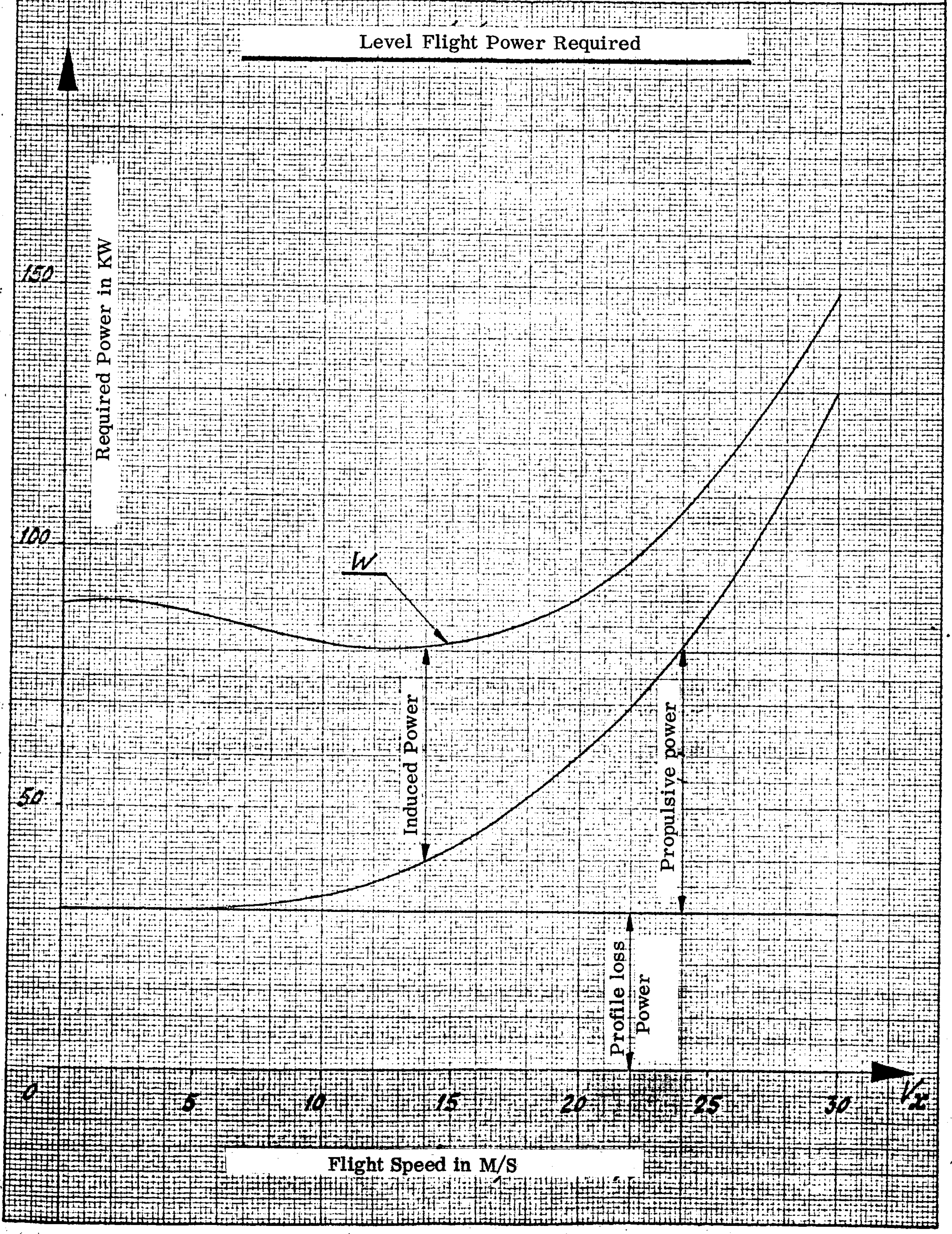
DP 03 - Z - 1108

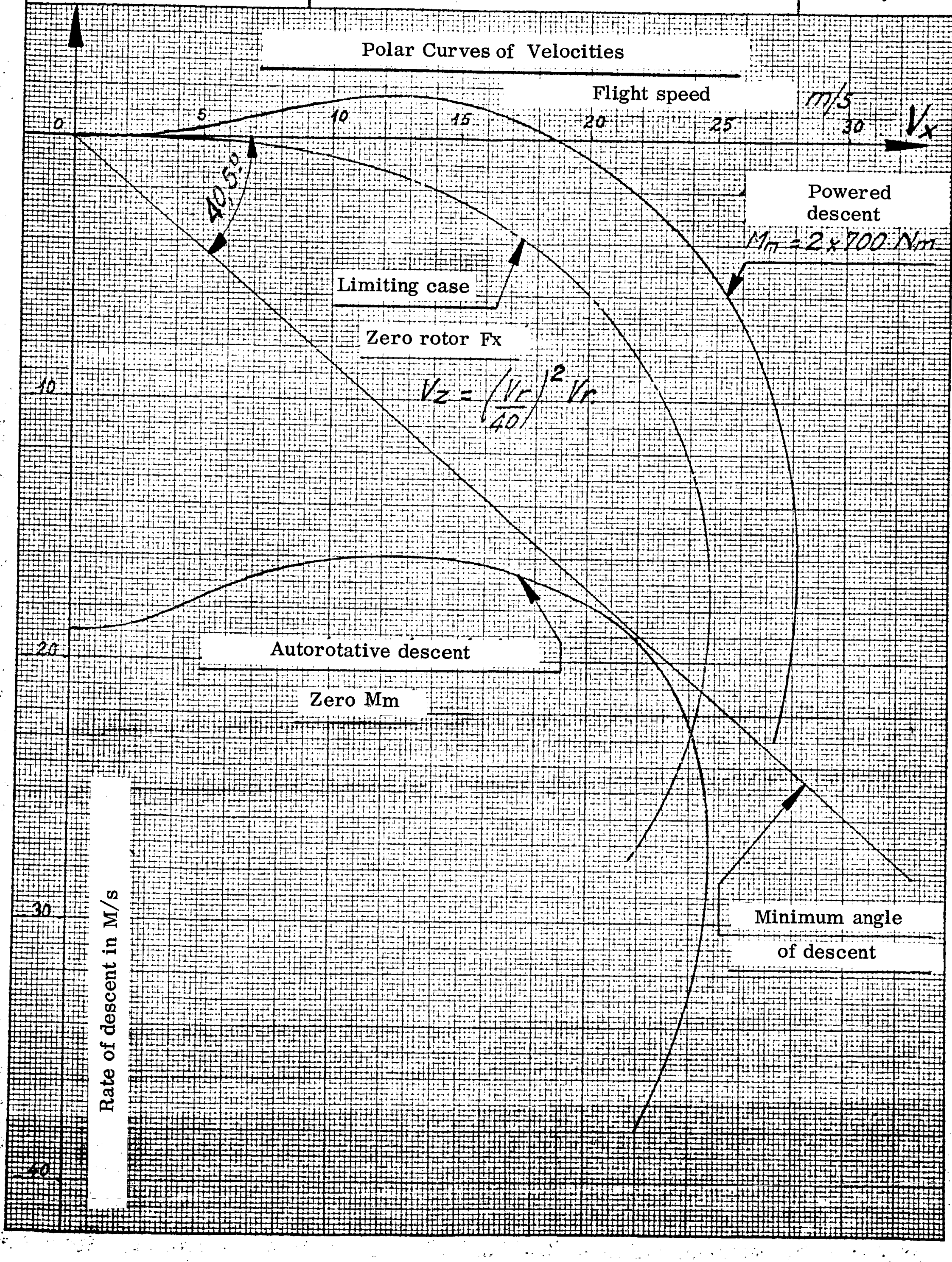
49-2

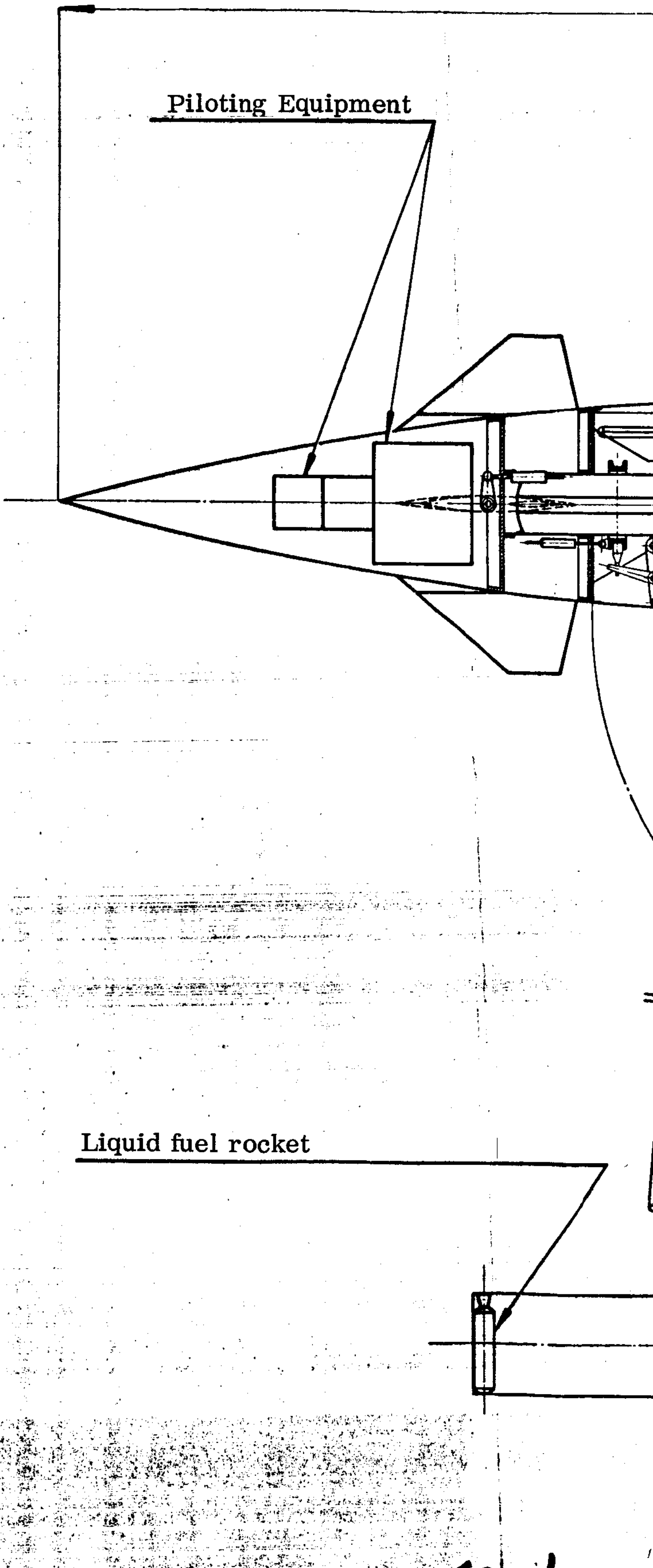
Power Required in Hovering

(Ground effect)

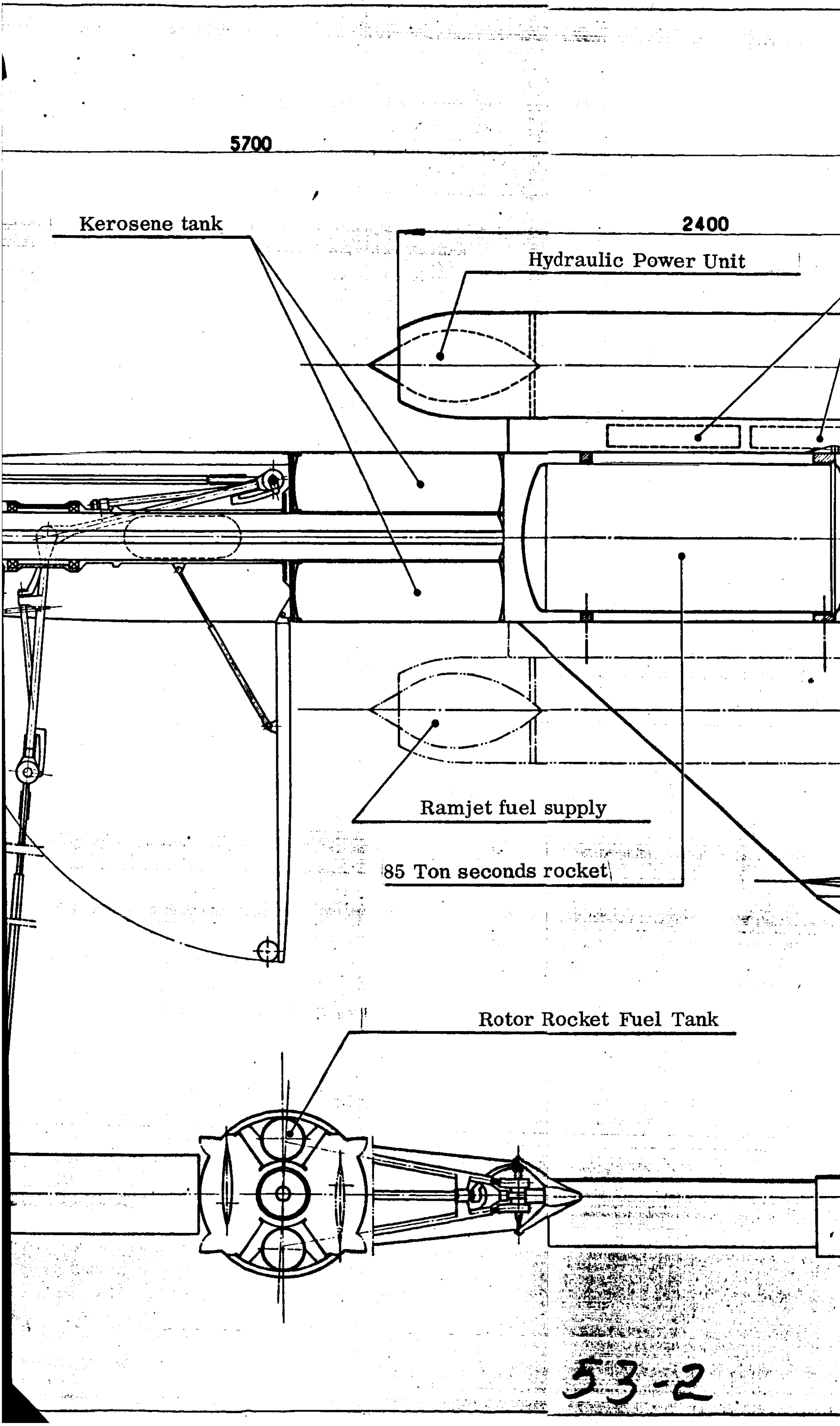



Level Flight Power Required
Required Power in KW
150
100
50
W
Induced Power
Propulsive power
Profile loss Power
0
5
10
15
20
25
30
Vx
Flight Speed in M/S

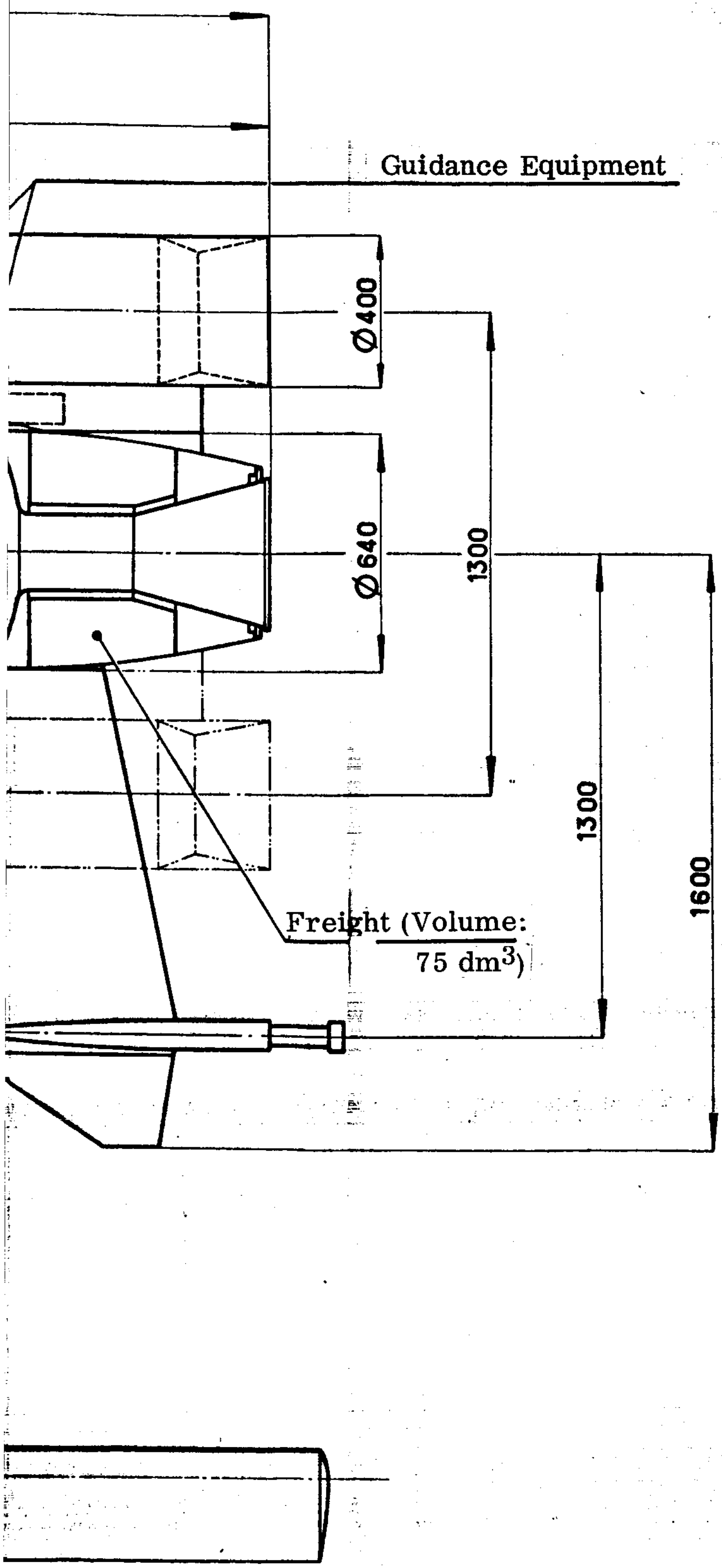

Polar Curves of Velocities
Flight speed
m/s
V_x
0
5
10
15
20
25
30
$40{,}5°$
Powered descent
$M_m = 2 \times 700$ Nm
Limiting case
Zero rotor Fx
$V_z = \left(\frac{V_r}{40}\right)^2 V_r$
10
Autorotative descent
20
Zero Mm
Minimum angle of descent
30
40
Rate of descent in M/s



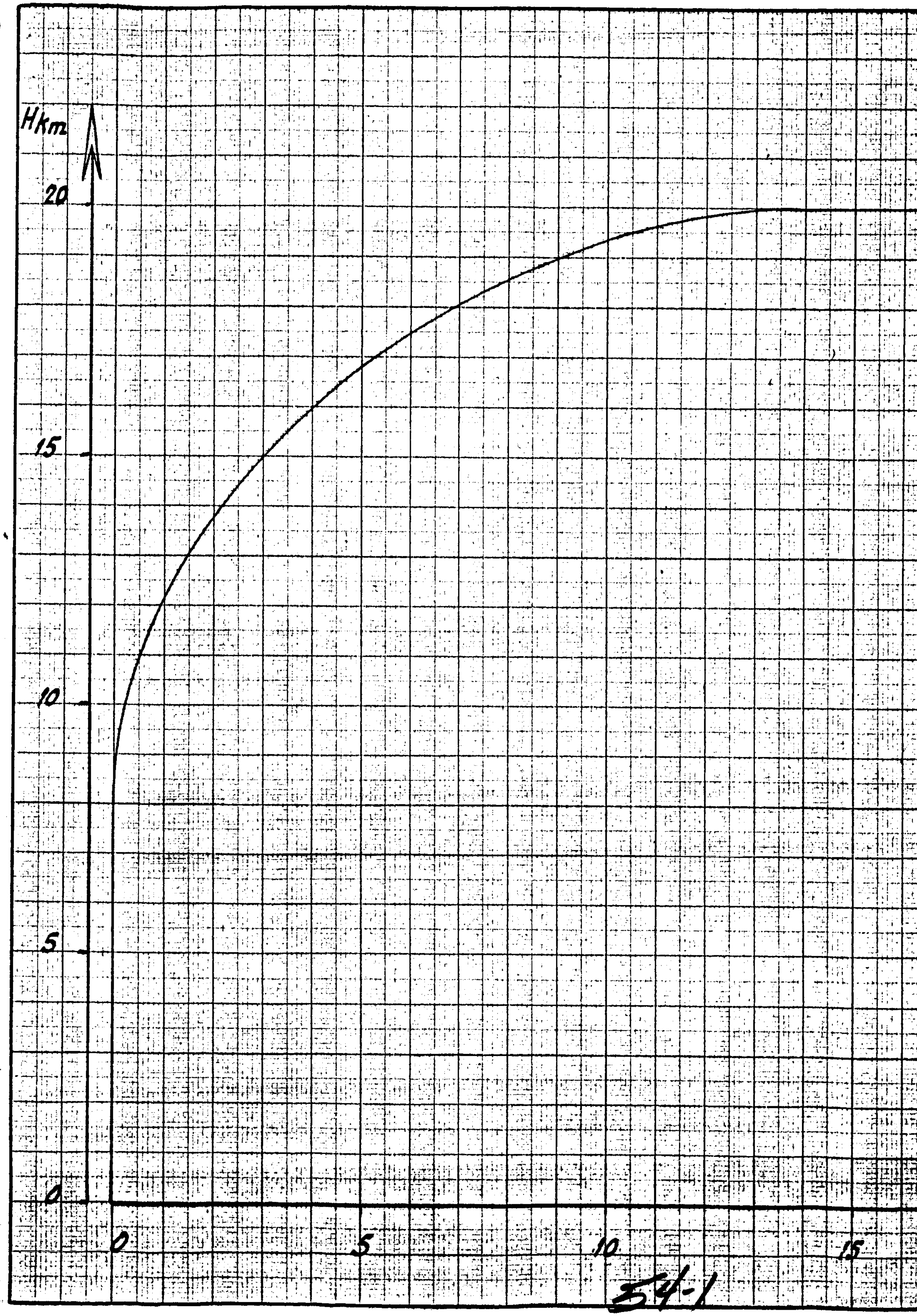
53-1


5700
Kerosene tank
2400
Hydraulic Power Unit
Ramjet fuel supply
85 Ton seconds rocket
Rotor Rocket Fuel Tank
53-2

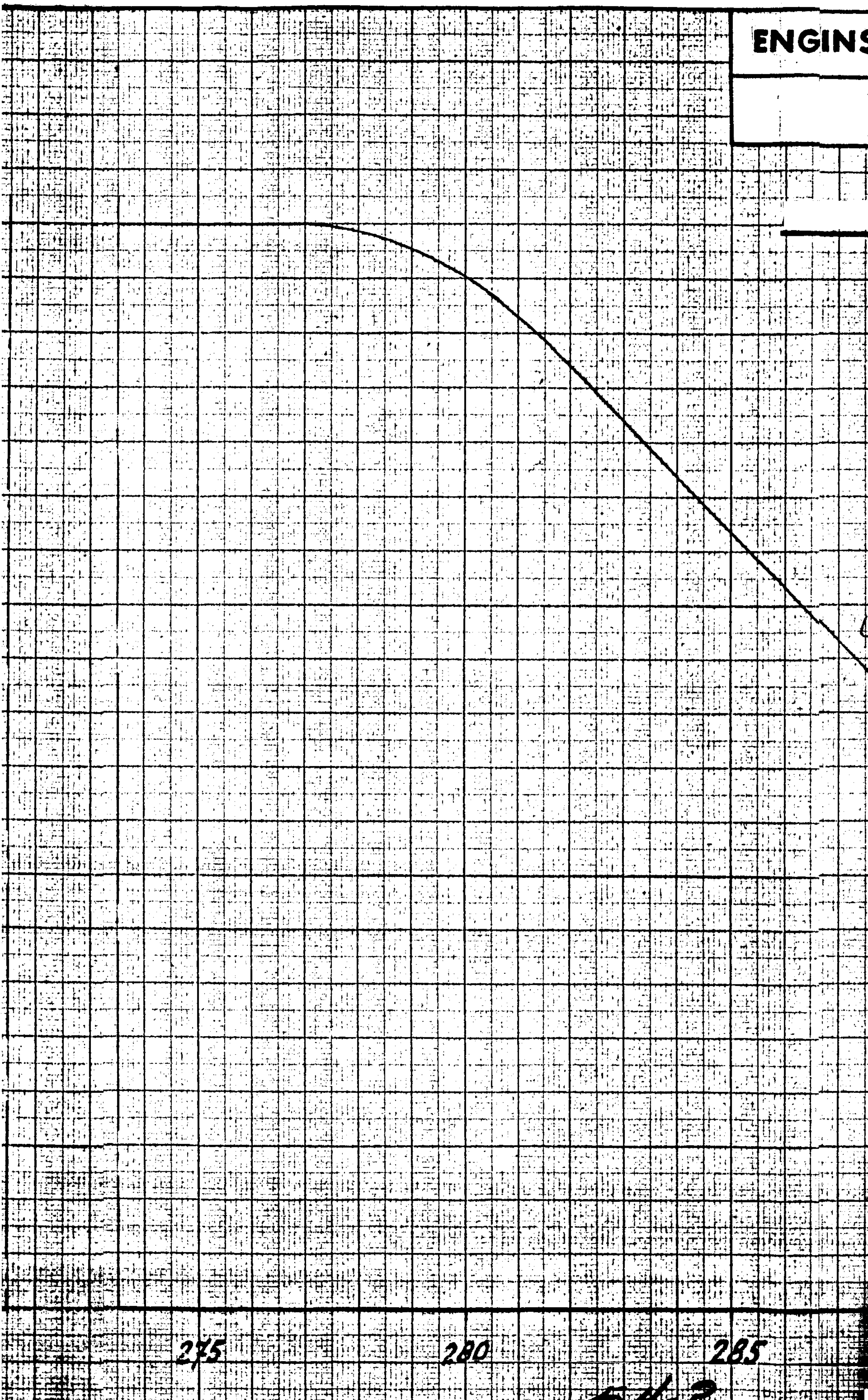
ENGINS MATRA	Postal Missile	-53-
	General arrangement of equipment	



53-3

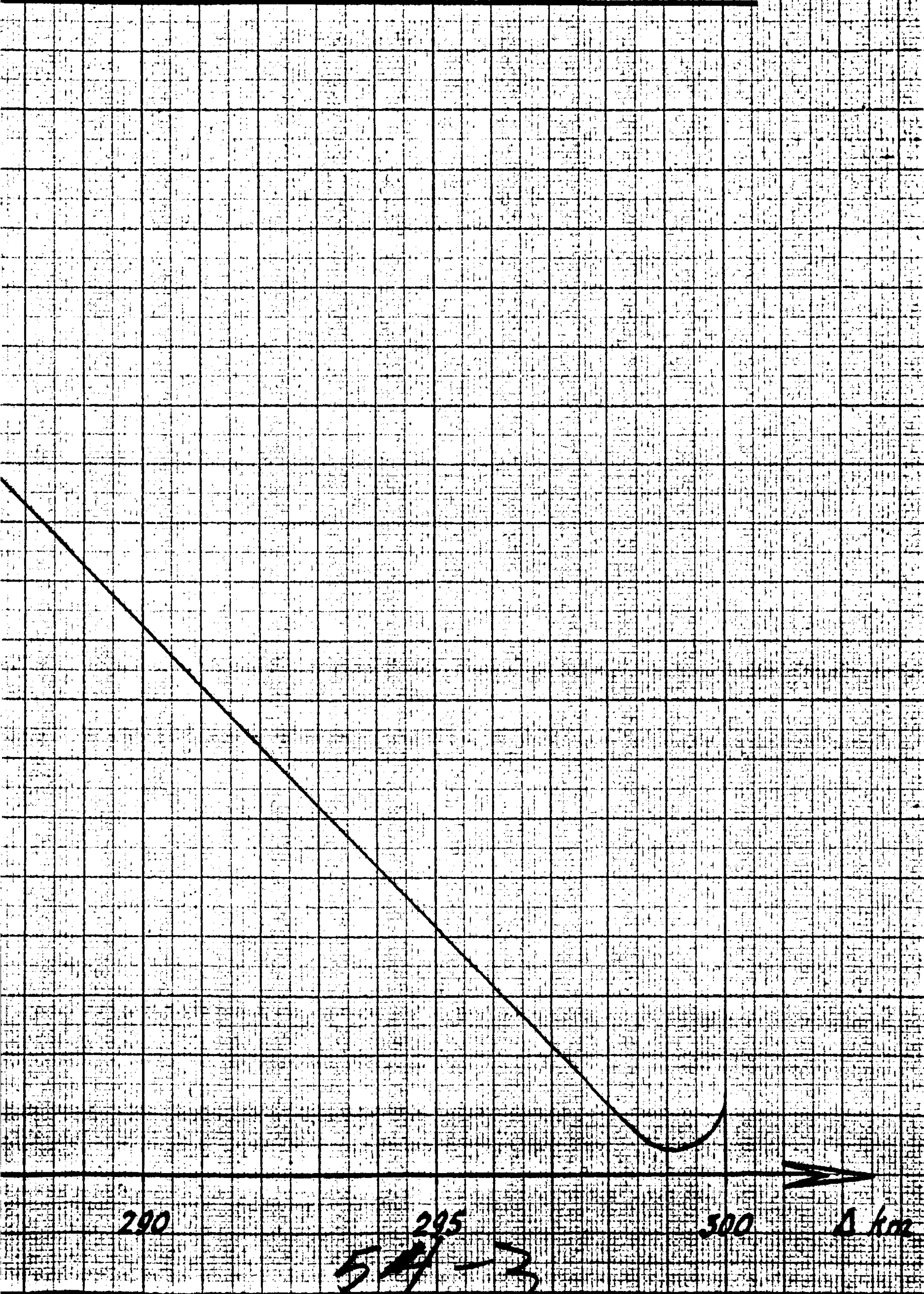


5-4-1

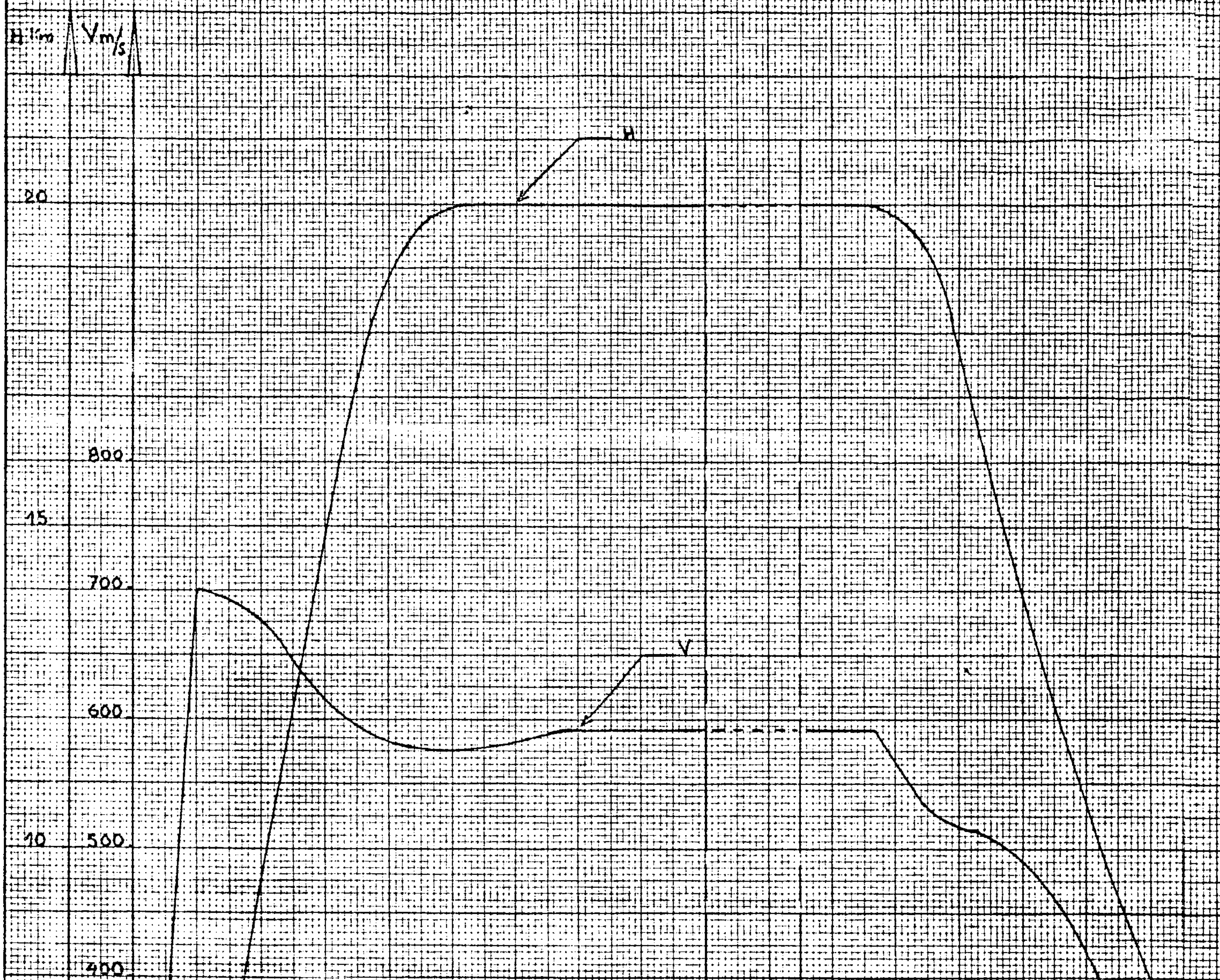


54-2

Altitude as a function of the traversed distance

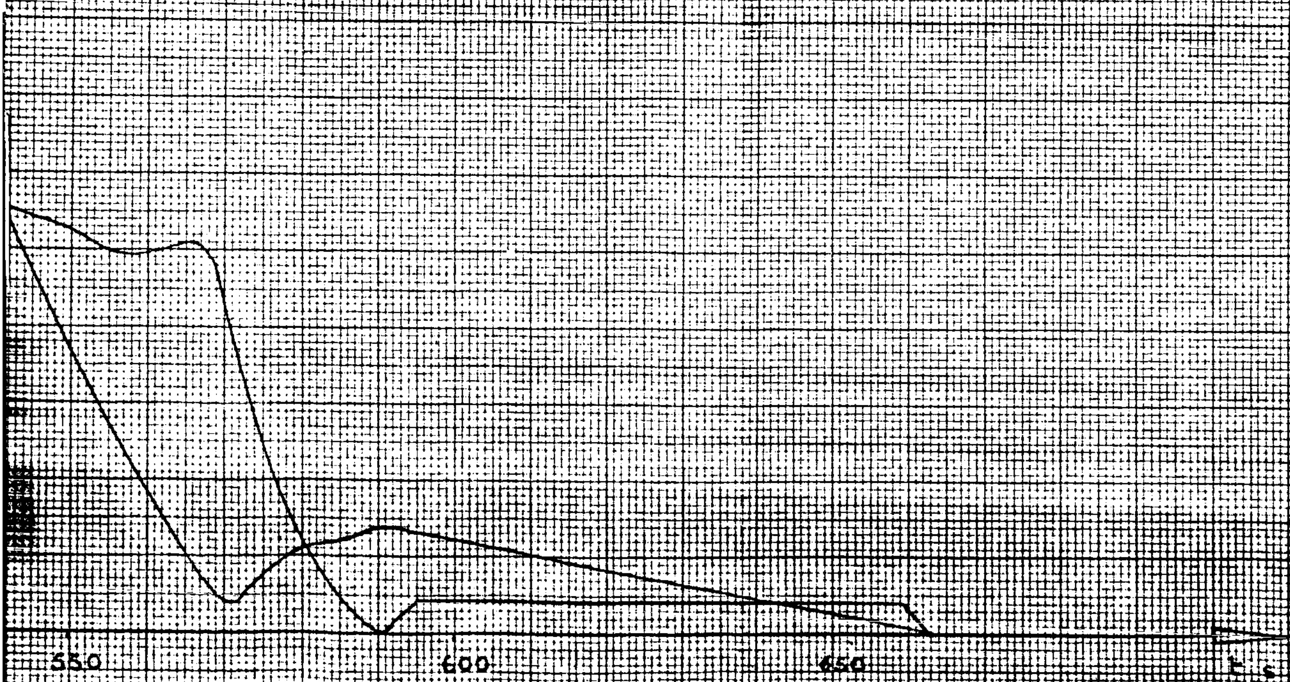


54-3



55-1

Altitude and speed as a function of time



55-2

CHAPTER 5

SPECIFICATION OF GUIDANCE

5 — 4.3 — Guidance mode.

5 — 4.4 — Missile Flight during the descent.

5 — 4.41 — Transition phase to start the autorotative descent.

5 — 4.42 — Autorotative descent.

5 — 4.43 — Pull-up.

5 — 1 — GENERAL

5 — 1.1 — Principle of Guidance

The theoretical trajectory of the missile center of gravity is a curve described in the plane defined by the center of the earth, the launching point 0 of the moving body and its arrival point A. This curve is shown in page 54.

In fact, the real trajectories will be warped curves. The guidance processes which we are about to define are aimed to maintain the real trajectories as near as possible to the theoretical trajectory by applying at every instance the appropriate forces to the moving body.

A trajectory can be divided into four successive phases: the climb, cruising, approach and descent. As we shall see in what follows, different guidance methods correspond to these different phases. We shall discuss one by one those envisaged for cruising, approach and descent.

In addition, for reasons of convenience of applying control signals to the various power units of the missile, we have chosen to resolve the guidance laws into certain preferred directions described as follows:

- During cruising, the local vertical and the line normal to the plane of the theoretical trajectory.

- During the approach, a (variable) direction in the plane of the theoretical trajectory and the normal to that plane.

- During the descent, the direction of the horizontal component of the wind and the perpendicular horizontal direction.

In general terms, the guidance laws proposed here are functions of the position of the center of gravity of the moving body, and of the orientation in relation to the earth of its longitudinal axis. The position is given by different localizing devices (one for cruising, and one for the descent). The orientation of the axis is obtained by using gyroscopes placed on board.

5 — 1.2 — Monitoring of the Missile.

To ensure that the above resolution of the guidance laws corresponds to a simple allotment of duties on board the missile it is necessary that the moving body should have a suitable orientation in relation to the ground. If so, the resolution carried out relative to directions linked with the earth is, in fact, identical with that carried out relative to the preferred directions of the missile.

We have chosen to perform the cruising and the approach in such a way that the yawing axis of the missile is substantially parallel to the plane of the theoretical trajectory. This control is obtained by acting on the aerodynamic rolling control surfaces.

During the descent, on the other hand, the plane of the theoretical trajectory no longer plays any special part. The control of orientation reserved for this phase consists in maintaining the pitching axis of the missile approximately horizontal. This result is obtained by a rotation of the missile about its yawing axis which, in itself, is inclined by less than 30° to the horizontal.

We shall develop the notion of the orientation of the missile in appendix I by defining the quantities which make it possible to fix this orientation.

5 — 1.3 — Airborne gyroscopes

We have stated under 5 — 1.1 that the missile is equipped with gyroscopes and we shall return to this problem in the course of Appendix I. Let us nevertheless state immediately that these gyroscopes are two in number so that four items of information are available on board which we can apply in the following manner:

- One, for the control in roll during the entire flight which we shall designate by e_1.
- One, (denoted by e_2 in what follows) for the control in pitch, also during the entire flight.
- One, for the control in yaw during the climb, cruising, and approach. This will be denoted by i_1.
- The last one, (being i_2) for the yawing control during the descent phase.

These notations, applied to these items of information, indicate that they constitute sensing signals in relation to the outside or inside hinges, respectively of the gyroscopes called No. 1 or No. 2.

5 — 1.4 — Flying Controls of the Missile

During the climb, cruising and approach phases, three classical linear control loops will be used to control the missile with the help of the aerodynamic control surfaces for roll, pitch and yaw.

Certain guidance principles in use rely on gyroscopic sensing signals which translate the attitude of the missile. It should be noted that a correct determination of the coefficients must then be envisaged within the framework of a complete analysis of stability, an analysis which includes that of rapid oscillations of incidence and of side slipping.

In view of the static stability margins which we have provided, the problem of adapting the different guidance and stabilization coefficients should present no major difficulties if each of the three control chains contains active phase advance elements, namely 3 rate gyros will be applied to ensure the correct damping of rapid oscillations about the center of gravity.

The analysis of guidance system oscillations which leads to an exact determination of the filters and the required correcting networks can be easily performed on an analogue computer.

During the phase of descent with unfolded rotor, three linear control chains are used for the control in pitch and yaw and of the vertical velocity carried out by means of the rotor.

An off-or-on control chain which controls the rolling control nozzles is envisaged in order to orientate the missile suitably in relation to the wind.

Let us note that the analysis of flight control during that phase will largely make use of preliminary wind tunnel tests and of full scale development in the portal frame test rig.

5 — 2 — CRUISING PHASE

5 — 2.1. Guidance Equipment.

We have previously mentioned under 2-2 the structure of the guidance loop of the missile under automatic remote control.

The localization is carried out by a LY Radar installed at the landing point of which the telementering part will be modified to ensure automatic following of the missile during its cruising flight. The missile, equipped with a super heterodyne responder such as the RS2, will be taken over by the radar in the course of the vertical climb.

The analogue computer used in the cruising phase is very simple. It provides the following functions:

- Transformation of the spherical radar coordinates into cartesian coordinates.

- Working out of control signals which will be very simply linked to the cartesian coordinates (see under 5 — 2.2, below).

These different operations require the application of about 10 amplifiers.

The transmission of guidance control signals to the missile is performed by classical remote control equipment. For cruising navigation, two linear transmission channels only are necessary (pitching and yawing). On the other hand, as we shall state in Sections 5 — 3 and 5 — 4, four continuous channels (G_1; G_2; G_3; G_4) and four intermittant signals (S_1; S_2; S_3; S_4) must be applied during the approach and descent phases.

The carrier frequency of this communication will be established at about 250 Mc/sec. A double frequency modulation such as that embodies in the S. F. E. N. A. equipment can be utilized.

The stage length of 300 km should be obtainable without major difficulty. It is, in fact, possible to provide the ground aerial with sufficiently pronounced directional properties. In addition, the high altitude of the missile facilitates the establishment of the communication.

5 — 2.2. — Guidance Laws.

- Roll.

The yaw axis of the missile is maintained parallel to the plane containing the theoretical trajectory in controlling the rolling maneuvers by the sensing displacement e_1 of the outside frame of gyroscope No. 1.

In this way, the allocation of the guidance control signals can be simply carried out. The deviations computed on the ground can be directly applied to the missile coordinate frame in pitch and yaw.

- Pitch.

The guidance error signal is:

$$\Delta Z = Z_{th} - Z$$

In order to ensure the correct control of the desired altitude it is necessary to introduce in the circuit a phase advance term.

The control signal computed on the ground will have the form:

$$k_1 \Delta Z + k_2 \frac{dZ}{dt}$$

It will also be possible to introduce in the servo-control, the sensing displacement e_2 of the outside frame of gyroscope No. 2.

Note:

This control signal will not be transmitted in this form unless the missile will have reached an altitude near the cruising altitude. The turn carried out during the climb phase will be obtained by remote transmission to the missile of a programmed control signal which makes it follow substantially the trajectory shown under 4 — 6 or else by providing this program on board the missile.

- Yaw

The ground computer will work out at each instant the distance Y of the missile in the vertical plane which contains the theoretical trajectory. In order to maintain in addition the longitudinal axis of the missile substantially parallel to this plane, the guidance control signal used will be:

$$k_3 Y - k_4 i_1$$

where i_1 is the displacement of the inside frame of gyroscope No. 1.

5 — 3 APPROACH PHASE

5 — 3.1 — Special Requirements.

We call "approach" that part of the trajectory during which the missile changes over from the cruising flight conditions to those of the unfolding of the rotor.

The initial conditions of this phase are therefore:

- Horizontal flight,
- Altitude of 20,000 m,
- Mach No. = 2.

By a diving maneuver followed by a tail-down maneuver the missile is bought into the final condition:

- Vertical flight climbing along the vertical line over the landing point.
- Altitude of 1200 m.
- Speed = 50 m/sec.

The special nature of the task consists in the fact that the pull-out must be carried out at slow speed (necessary conditions to ensure that the rotor unfolds correctly) and that the missile is then difficult to handle. It is therefore desirable to avoid starting the pull-out except under the best possible conditions and above all to avoid a large scatter of the initial conditions of flight between one missile and another.

We envisage achieving this aim by applying an intermediate guidance during the dive such that the position and orientation of the missile at the beginning of the pull-out should substantially always be the same. This guidance makes it possible to obtain arcs of trajectory which are very near each other, while the missile speed at the end of this dive will also be subject to a small scatter only. In addition, in order to facilitate the control of the missile at low velocity we carry out this pull-out maneuver at the lowest possible altitude.

5 — 3.2 — Guidance Equipment.

The guidance of the missile is, once again, carried out by automatic remote control during this phase.

As soon as the distance of the missile from the landing point reaches 30 km, a total LB radar searches to lock on with the missile in direct communication. It will be possible to help the LB radar in making use of the localization of the missile by the LY Radar through a responder link.

In fact, it may be feared that the LB Radar may be unable to lock on the missile safely before the beginning of the approach phase.

It is more advisable to envisage the change of localizer during the approach phase only after the pursuit of LB Radar is functioning correctly. This maneuver can be carried out without any noticeable repercussions on the control signals for guidance.

The analogue computer used will be substantially more complex than the cruising computer. In fact, as we shall see in the following sections, the computation of the control signals requires a larger number of computing operations. In Appendix II we shall present the different functions of the computer for approach and descent with unfolded rotor as well as its working diagram. We may mention already that the required computer equipment will essentially consist of 20 amplifiers, 3 servo-motors and several transducers.

The remote control equipment described earlier for cruising will also be used for the approach. During this phase two continuous channels (G_1, G_2) and two on-or-off signals (S_1, S_2) will be used.

5 — 3.3 — Dive.

The dive is released from the ground by signal S_1 which is emitted when the missile is at a distance of about 25 km from the landing point.

As soon as this signal is received, the missile dives by taking up the maximum permissible incidence until its longitudinal axis makes an angle of about 45° with the plane of the horizon.

From this instant onwards, the missile follows the trajectory aimed to take it to the neighborhood of a certain point in the plane of the theoretical trajectory (altitude, about 750 m, distance from the point of landing, about 1600 m). The missile has, its longitudinal axis still at an angle of about 45° against the horizon.

The rolling control loop ensures the servo-control link described under 5 — 1.2 (yaw axis parallel to the plane of the theoretical trajectory) by making use of the outside frame sensing displacement of gyroscope No. 1.

The pitching evolution is obtained by setting the corresponding aerodynamic control surfaces. The control consists of the following:

- First, of a voltage supplied by an airborne source (maximum dive),

- Subsequently, by the difference between the gyroscopic sensing displacement e_2 and the output signal G_1 of the receiver on board, (controlled dive).

The quantity transmitted by channel G_1 is computed automatically on the ground as a function of the position of the moving body at every instant and of the theoretical trajectory which the body should follow. The formulation of this quantity is given in Appendix II.

The changeover from the maximum dive to the controlled dive is carried out by a commutation of the inputs of the flight control chain in pitch controlled by gyroscope No. 2.

During the entire dive, the yaw control chain is controlled by a combination of the inside frame sensing displacement of gyroscope No. 1 and of the output G_2 of the receiver on board (deviation computed on the ground, of the moving body from the plane of the theoretical trajectory).

This yawing control has the effect of bringing, and later maintaining, the moving body into the neighborhood of the plane of the theoretical trajectory while the longitudinal missile axis remains substantially parallel to this plane. The mathematical formulation of this control signal is given in Appendix II.

5 — 3.4 — Nose lift maneuver.

During the nose-lift, we endeavor always to maintain the missile within the plane of the theoretical trajectory namely with its pitching plane almost parallel to that reference plane.

The control signals applied at the inputs of the roll and yaw control chains are therefore the same as during the dive.

The pitch evolution is obtained simply by modifying the computation of the quantity transmitted by the channel G_1. This modification is automatically embodied by the ground equipment when the altitude of the moving body is 750 meters. Appendix II formulates this commutation and the new form of G1.

In fact, a second switching operation is carried out on the ground through the transmission channels G_1 and G_2 when the missile is at a small horizontal distance (about 250 meters) from the landing point (appendix II also deals with this point).

The reason for this further modification is that, in order to carry out finally the unfolding of the rotor, it is important that the longitudinal axis of the missile should be as near as possible to the vertical (within a cone with a apex semi-angle of 20°). This condition is related to the intention to carry out the unfolding at the exact vertical over the point of landing.

Some calculations have been made on the assumption of large initial deviations or in taking account of the wind. These calculations lead us to estimate that the horizontal deviation at the instant of unfolding must not exceed 200 meters.

5 — 4 — PHASE OF DESCENT WITH UNFOLDED ROTOR.

5 — 4.1 — Introduction.

We have indicated earlier, under 4—6, the three main phases which make up the descent.

- The first phase, which from the unfolding of the rotor and its starting up is designed to ensure control of the missile while the vertical velocity diminishes and then changes sign.

- The second phase, using the rotor in autorotation, allows the missile to carry out a stabilized descent at a constant speed without adding power.

- During the third phase, released near the ground, the rotor receives the energy of its power device. The missile is subjected to a vertical thrust at a load factor of 1.5, which vertually cancels its rate of descent at touch-down.

During the second phase of stabilized descent it is essential that the missile catches up on any horizontal deviation which exists at the end of the approach. It is therefore within the framework of this second phase that we should define the horizontal guidance of the missile.

In the first place, it is appropriate to formulate the possibilities of the flight evolution of the missile under the conditions of stabilized descent when analyzing the different possible configurations (5 — 4.2). We shall later present the guidance principles utilized which are compatible with these possibilities (5 — 4.3).

Finally, a description of the evolution of the missile in the course of the three descent phases will be given (5 — 4.4).

5 — 4.2 — Possible Missile Maneuvers.

The descent in autorotation can be carried out following two essentially different flight configurations.

- Classical descent, called "into wind" during which the missile axis is included in the direction of the transverse movement. It is the horizontal component of the sustentation force of the rotor which produces the movement.

- A special descent called "axial flow" during which the axis of the missile is inclined in a direction opposite to the transverse movement. In this case it is the horizontal component of the aerodynamic force acting on the missile which produces the movement.

Considering the duration of this phase and the small magnitude of the deviation to be restored, the translational speed of the missile requires for correct guidance can be small (a few meters per second). However, it is appropriate to consider the horizontal wind which can attain much greater speeds.

Two possibilities appear:

- The missile carries out an oblique descent, the slope of which is determined by the mean wind velocity. The missile maneuvers have the sole purpose of retrieving the initial deviations and to counteract the gusts.

- The missile carries out a vertical descent during which it must simultaneously restore the initial deviations and overcome the aerodynamic forces due to the wind. The missile must therefore be able to take up, in relation to the wind a translational velocity at least equal to that of the wind (more accurately, exceeding the latter by a few metres per second).

Within the framework of the vertical descent we have made the comparison of the two flight plans of which the missile is capable. Page 75 shows the equilibrium states of the missile under the two conditions for translational speeds between 0 and 20 m/sec in assuming the aerodynamic center and center of gravity to coincide.

Descent into Wind.

In the case of zero wind, the minimum vertical velocity is 19 m/sec. When the horizontal velocity increases, the missile tilts. The vertical velocity then slightly diminishes, only to increase rapidly subsequently while the attitude and the incidence increase. Beyond 20 m/sec, the tilt which the missile must take up in order to balance the aerodynamic forces becomes prohibitive. This is essentially due to the large sizes of the frontal area of the missile of which the SC_x is about 5 m^2.

During this descent, the rotor operates as the rotor of an autogyro or of a helicopter in power-off descent. It is approached by the flow from below in order to drive the rotating wing in autorotation.

Axial flow descent.

In this solution, the flow through the rotor is practically axial whatever the wind velocity.

This descent is distinguished by high rates of descent (50 m/sec) and permit the attainment of translational speeds exceeding 30 m/sec.

For these two descents it is attempted to maintain the aerodynamic center as near as possible to the center of gravity in order to avoid the appearance of drag forces on the rotor. Having regard to a larger relative wind, the second solution is more sensitive to the position of the aerodynamic center of the missile.

In the particular case of the missile presented here, the position of the aerodynamic center at small incidences is much further out than at large incidences (see p. 47), in fact, the shift is 1 meter.

In relation to the center of gravity at the end of the flight, the distance between the aerodynamic center and the center of gravity is of the order of 1.1 m for the second mode of descent while the distance is practically zero for the first mode at very large incidences.

In order to make the axial flow flight maneuver acceptable, it would be necessary to provide moving or inflatable surfaces on the missile which correct, during the descent phase, the position of the aerodynamic center. It would also be possible to use a three-bladed rotor with offset hinges in order to introduce large moments capable of balancing the missile in spite of the low position of its aerodynamic center. All the same, this latter solution includes a big mechanical complication which substantially increases the manufacturing cost of the rotor.

All told, we shall adopt the classical mode of flight of the missile "into wind". For wind speeds below 60 km/hr, we can perform a vertical descent. For higher wind speeds it is necessary to apply a sloping path of descent determined by the mean wind speed.

5 — 4.3 — Guidance Mode.

The guidance consists in maintaining the missile along a path of descent either vertical or sloping.

The chosen navigation law will therefore be that of track holding.

In the experimental stage, we propose to carry out the guidance under automatic remote control making use of equipment also applied in the approach phase.

- LB radar for localization,

- Analogue computer set up to compute control signals during the descent,

- Remote control equipment.

The essential requirement is that of precision.

In fact, the precision of total radar in normal service is of the order of:

10 m in distance,

1 milliradian in bearing and elevation.

The values are too large to ensure the landing accuracy which we expect.

It can, however, be noted that, provided the radar is installed at a small distance from the landing ground, (a few hundreds of meters, for example), the measuring error due to the servomotors in bearing and elevation is negligible compared with the distance error. In addition, the computer can work out the deviations relative to a vertical axis or a sloping line of descent. In the following sections and in Appendix II which deals with the computer, we envisage solely the first mode of descent in order to avoid an excessive length in this report.

It should be well understood that this slope is a correction, an adjustment for the purpose of facilitating the guidance. If this slope were not applied, the missile would drift in the wind until a deviation appeared in relation to the path of descent. The guidance control signal which would result would tilt the missile axis so as to allow it to move against the wind. The eventual analysis of the precision of guidance will permit to evaluate the possible benefit in such a sloping line of descent.

The exact knowledge of wind during the descent is not therefore necessary. It should be sufficient, in particular, to measure the wind near the ground by a simple anemometric device. On touching down, that is to say at the instant when we require all the precision of guidance, the inclination of the missile will be correct.

It is, therefore, appropriate to proceed as follows:

- To orientate the missile in roll so that the wind direction is contained in the pitching plane.

- To tilt the missile in pitch by an angle which is a function of the wind speed.

To do so, it must be ensured that:

- The remote control channel G_3 will transmit the wind direction.

In setting this angle against the sensing displacement of the outside frame of gyroscope No. 1, the error signal of the rolling control chain will be defined.

- The remote control channel G_1 will transmit the attitude angle which the missile must take up.

The pitching maneuver of the missile will be controlled by the comparison of this control signal with the sensing displacement e_2 of the outside frame of gyrscope No. 2.

In order to ensure that these two operations are significant, it is appropriate, as we have said before, to impose on the missile co-ordinate frame a supplementary condition by which it is linked with the ground coordinate frame. During the descent, the pitching axis must not greatly depart from the horizontal. This condition will be met by maintaining near zero the sensing displacement of the inside frame of gyroscope No. 2 by the yawing control chain.

The guidance control signals in pitch and yaw will be transmitted by the channels G_1 and G_2, respectively, and superimposed on the signals which we are about to formulate. The deviation to be corrected will not introduce control signals in yaw requiring a tilt exceeding a few degrees. This remains compatible with a "loose" regulation of the pitching axis towards the horizontal.

5 — 4.4 — Missile Flight During the Descent.

We give here a general description of the different phases of this maneuver. The precise details concerning the operation of the automatic pilot are provided in appendix III.

5 — 4.41 — Transition Phase to Start the Autorotative Descent.

The unfolding of the rotor is controlled from the ground when the rate of climb falls to 50 m/sec. At this point the altitude is about 1200 m. At the same time, the roll orientation is performed with the help of the nozzle control device.

The rotor unfolding sequence includes the following operations (plate on p. 76):

1) Unfolding of the propulsive arms of the rotor which support the two external elements of the folded blade.

2) Lighting up of the rockets.

3) Emergence of the telescopic element of the blade restrained by the cable paid out through an irreversible reduction gear at the rate of 3 m/sec.

4) Unlocking of the blade arm when the centrifugal force exceeds a predetermined value.

5) Unfolding of the rotor restrained by the opening cable.

The rotor becomes effective one second after the lighting up of the rocket.

During the moments which follow, the climbing speed of the missile diminishes under the action of gravity, reduced by the growing lift of the rotor. In fact, the rotor opens up with a positive setting to provide a lift of the order of 2,000-3,000 N. In this way, the missile is stabilized naturally with a weak interference of the automatic pilot, sufficient to control the attitude. The completion takes place 8 seconds after the lighting up of the rocket and the missile performs during the subsequent four seconds an accelerated descent which takes it to a rate of descent of 20 m/sec.

During the entire duration of this transient evolution, namely during 12 seconds, the missile performs a change of attitude in pitch which is programmed as a function of the wind speed. In addition, it is possible to envisage a progressive superposition of guidance control signals in pitch and yaw during this phase. The rotational speed of the rotor is controlled by the collective pitch, regulated to follow a program of pre-arranged variation. Let us note that the duration of operation of the rockets is limited to 10 seconds, the time required for the establishment of flow permitting an autorotative descent.

5 — 4.42 — Autorotative descent.

The rotor rockets are stopped and, for a wind not exceeding 15 m/sec, an autorotative vertical descent in the form of flight into wind at a vertical rate of descent of 20 m/sec is envisaged. The roll orientation is defined by the wind direction and the trimmed attitude in pitch is defined by the wind speed.

Furthermore, the missile continuously responds to flight control signals in pitch and yaw, worked out on the ground by the guidance computer so as to carry out the alignment on the vertical line over the landing platform.

The autorotative descent lasts for about 65 seconds. It is stopped at a given altitude of the order of 70 m.

5 — 4.43 — Pull-up.

The autorotative descent is followed by a pull-up maneuver performed at a mean load factor of 1.4. This maneuver will start by a very fast recovery of the attitude in pitch by which a new equilibrium is reached, to a minimum in order to reduce the drift due to wind. (the drift is estimated as 2 m in the most unfavorable case).

This short phase, essentially transient, is followed by a pseudo-stabilized phase during which the missile decelerates under the effect of the increase in the rotor lift. The speed reduction is accompanied by a slow variation of the attitude in pitch as introduced as a function of the wind speed. This variation will be so programmed that the equilibrium of the horizontal aerodynamic forces is permanently maintained. (The horizontal component of the rotor thrust and the horizontal component of the aerodynamic resultant force on the body and the wings of the missile). The guidance control signals which are interrupted during the attitude recovery phase, are re-established during the pseudo-stabilized phase. In p. 77 we illustrate the evolution of the different variables distinguishing the motion in the case of a pull-up performed in a wind of 15 m/sec.

The pull-up maneuver, which is used to bring the missile down to the ground with a speed which is substantially zero but provided with increased maneuverability as a result of an increased rotor lift, will be completely automated.

In particular, the control of the vertical handling during the pseudo-stabilized phase, performed by the collective pitch, will make use of an error signal computed on the ground and transmitted by the linear remote control channel G_4.

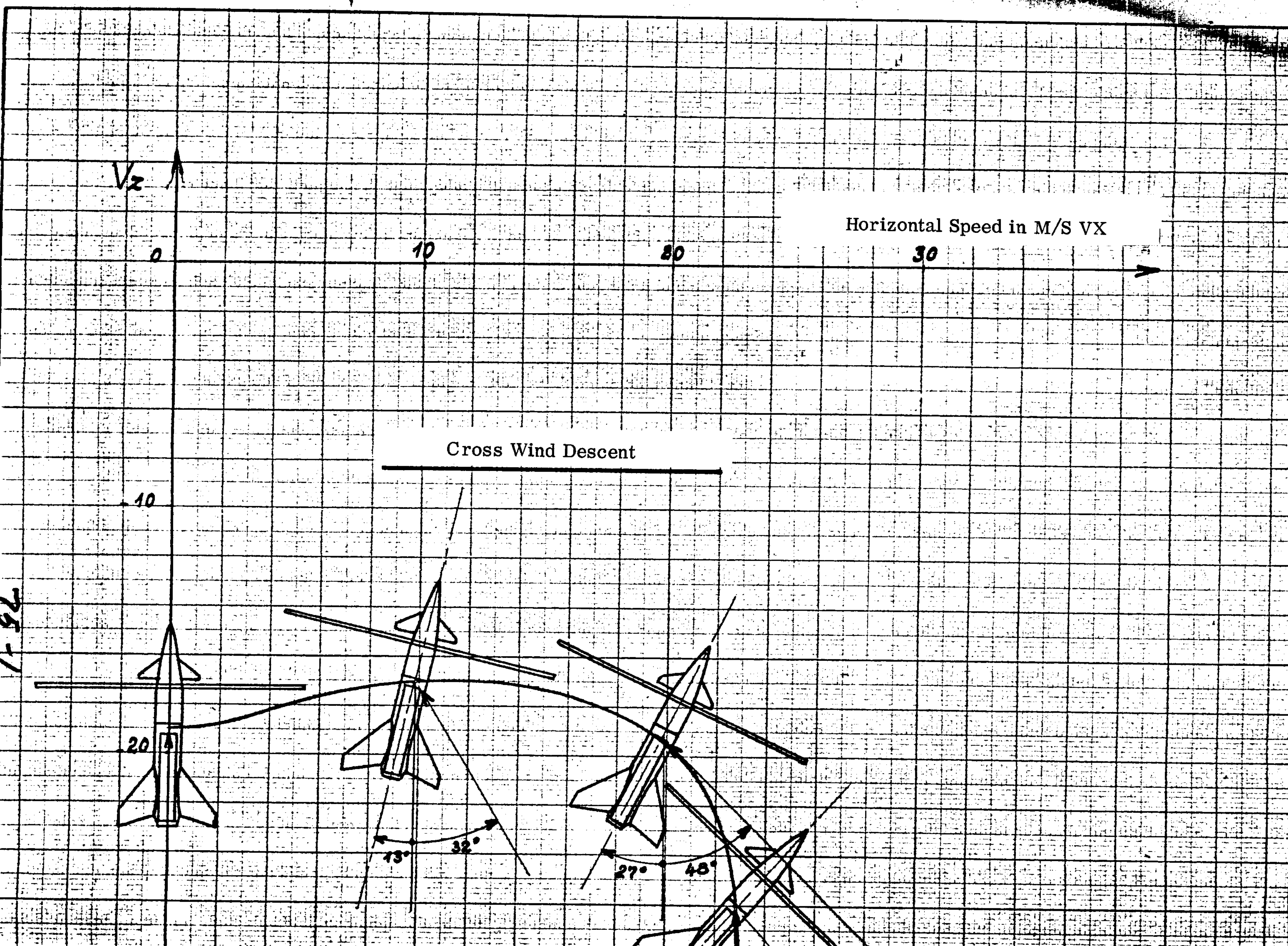
Let us emphasize that the landing procedure, whether with or without wind, will be developed in complete safety by experimenting in the portal frame (an experimental technique used in the development of the flying ATAR), where the missile will be maneuvered under conditions very near to the actual flight but provided with a cable which restrains the missile in case of a fault or a wrong maneuver.

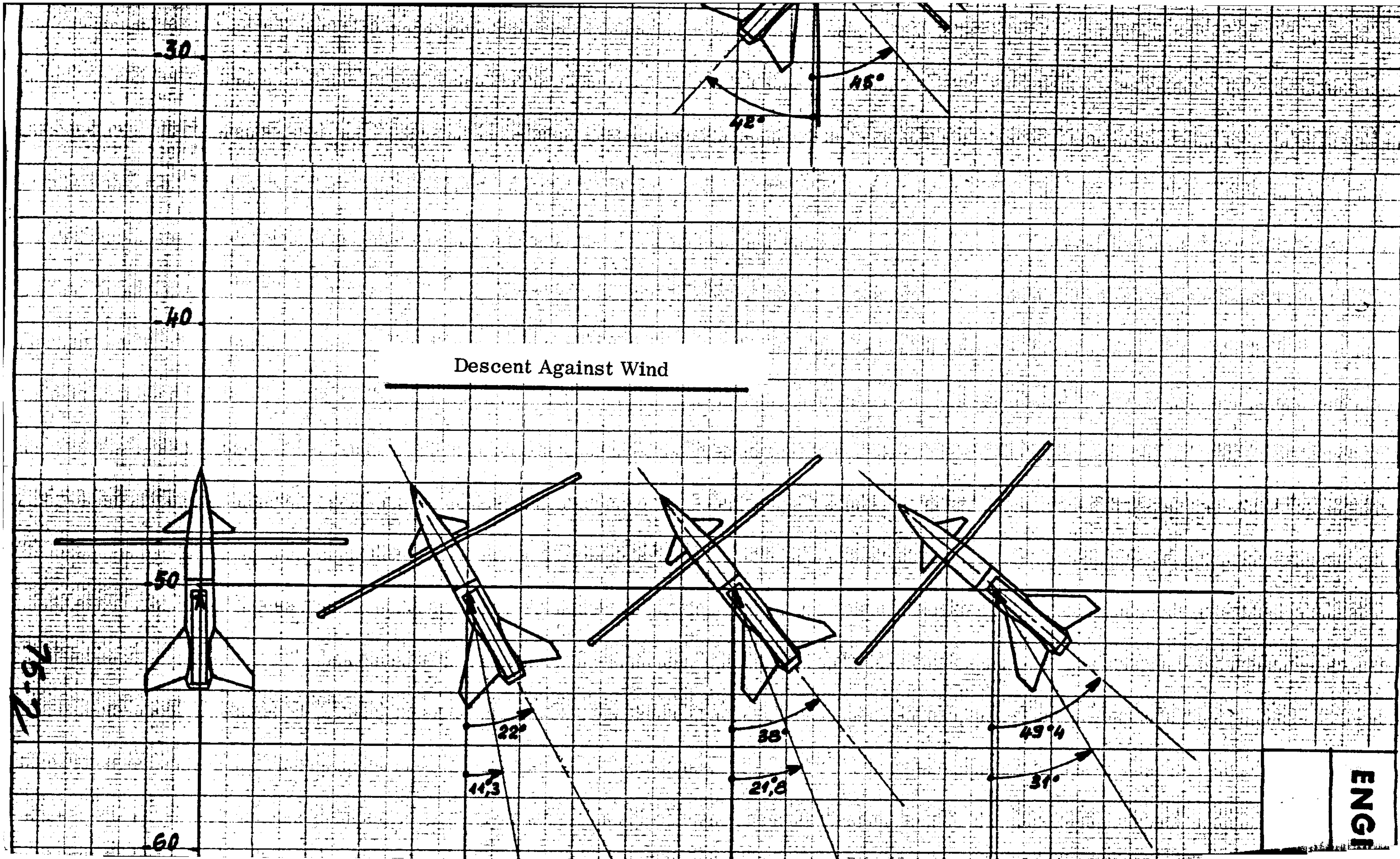
At touchdown, several precautions will be taken in order to ensure the safety of the machine:

- Lashing down of the missile.
- Stopping the rotor.
- Folding of the blades.

The lashing down system of the missile will ultimately be the object of a separate project study.

With regard to the rotor, the rockets will be stopped and in using a rotor brake, and a cable system actuated by a power take-off, the blades will be automatically folded and returned to their initial positions.


Cross Wind Descent
Horizontal Speed in M/S VX
Vz
0
10
20
30
10
20
13°
32°
27°
48°

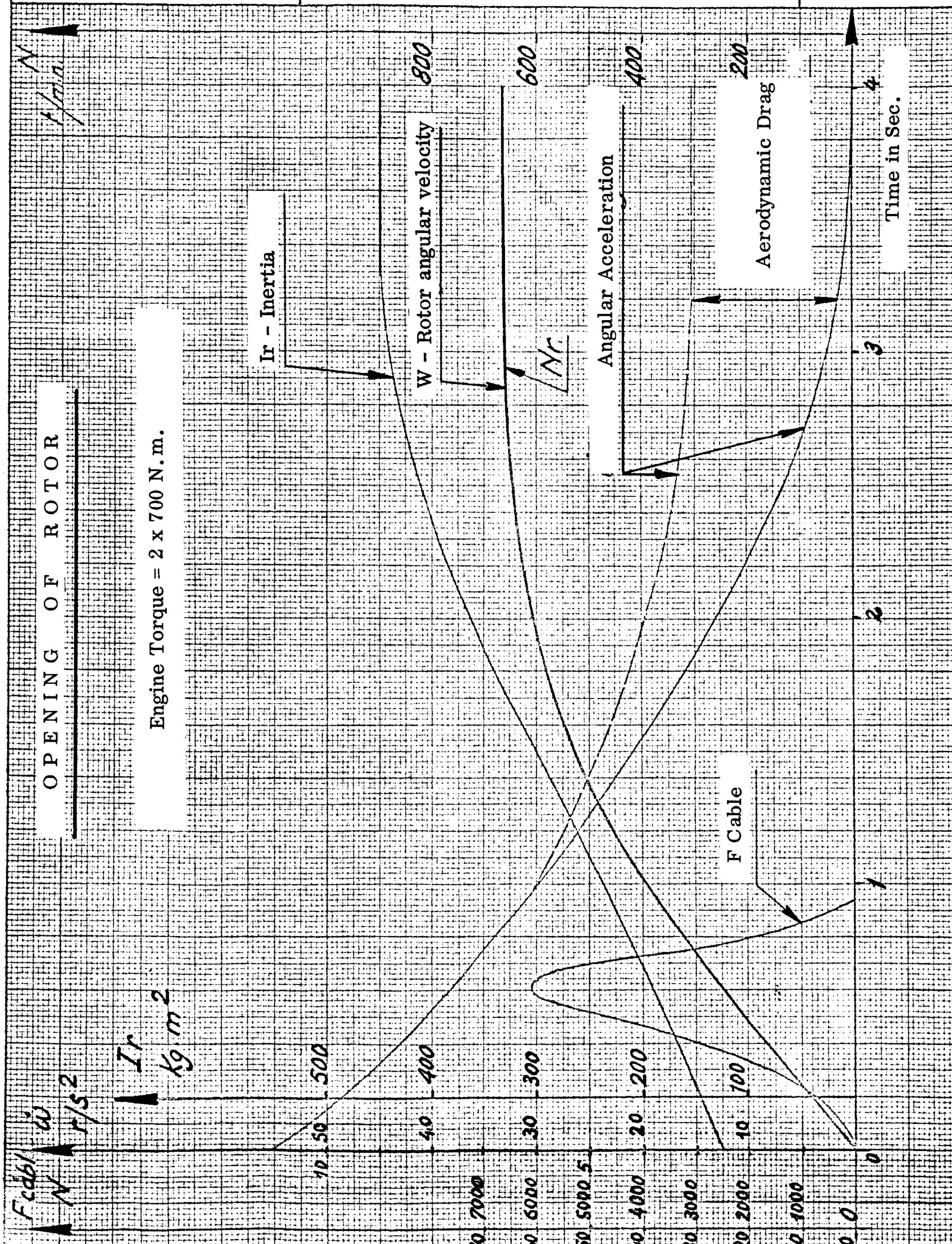


30
40
50
60
Descent Against Wind
45°
42°
22°
14,3
38°
21,8
49°4
31°
ENG
75-2

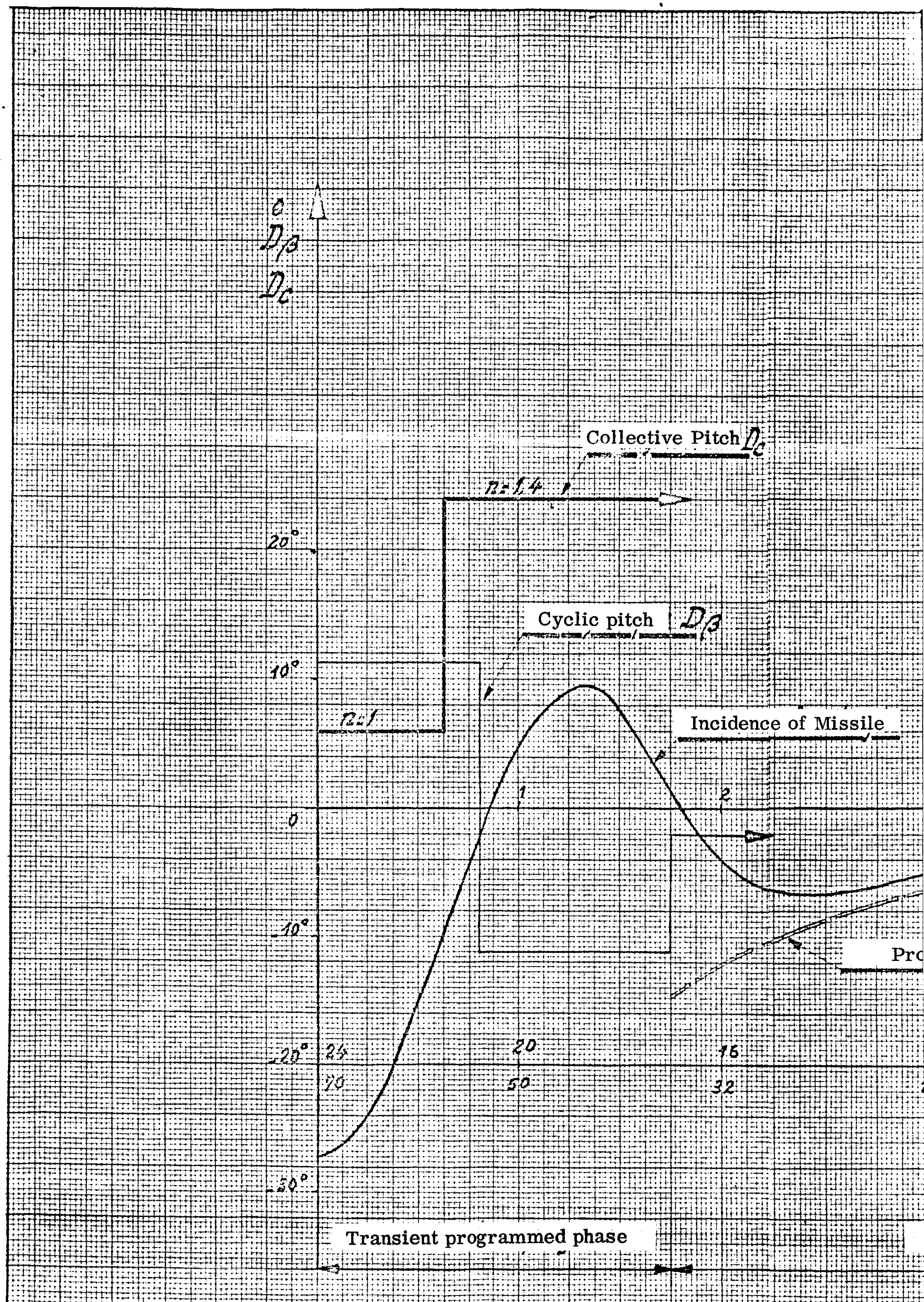
Missile Configuration During

Vertical Descent Engines Cut

Vertical Speed in M/S

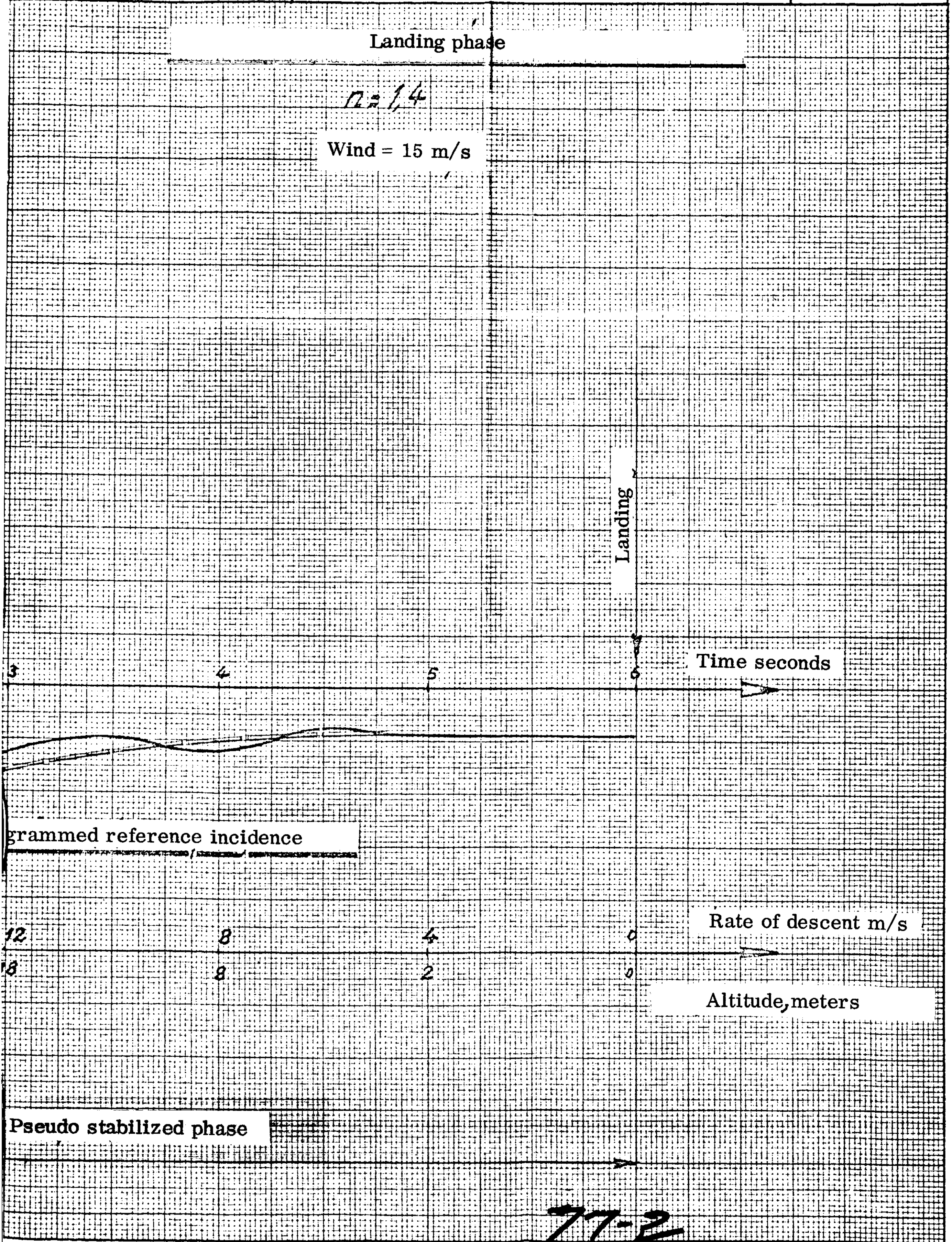


OPENING OF ROTOR
Engine Torque = 2 x 700 N.m.
Ir - Inertia
W - Rotor angular velocity
Nr
Angular Acceleration
Aerodynamic Drag
F Cable
Time in Sec.
N t/mn
800
600
400
200
Ir kg m2
ω̇ r/s2
F cabl N
ω r/s
500
400
300
200
100
0
10 50
40
30
20
10
0
7000
6000
5000 5
4000
3000
2000
1000
0 0
70
60
50
40
30
20
10
0 0
1
2
3
4



D_β
D_c
Collective Pitch D_c
n=1.4
20°
Cyclic pitch D_β
10°
n=1
Incidence of Missile
1
2
0
-10°
Pro
-20°
24
70
20
50
16
32
-30°
Transient programmed phase

77-1



Landing phase
n=1,4
Wind = 15 m/s
Landing
Time seconds
3
4
5
6
grammed reference incidence
Rate of descent m/s
12
8
4
0
Altitude, meters
8
2
0
Pseudo stabilized phase
77-2

APPENDIX I

ORIENTATION OF MISSILE WITH RESPECT TO GROUND

I - 1 - TRIHEDRA FIXED TO THE GROUND

I - 1.1 - Principal trihedron and secondary trihedron at landing point.

We consider as the principal trihedron fixed to the ground, a trihedron whose origin is at the desired landing point A and whose unit vectors are:

- $\vec{Z}$ along the downward vertical
- $\vec{X}$ horizontal and in the direction pointing away from the starting point 0.
- $\vec{Y}$ horizontal and such that (formula) is a trirectangular trihedron.

This trihedron is labelled "principal" because the theoretical trajectory is completely contained in plane $\vec{Z}\ \vec{X}$; direction $\vec{Y}$ is one of the prime guiding directions during the cruise and the approach.

We also use a secondary trihedron (formula) with the same origin A, obtained from the previous one by a rotation of algebraic magnitude about the oriented vertical $\vec{Z}$. The quantity δ is the angle (measured in the counter-clockwise direction of plane $\vec{X}\ \vec{Y}$) formed by the direction $\vec{W}$ of the horizontal component of the wind at A and direction $\vec{X}$.

This latter reference is used during the downward phase; δ is conventionally chosen as zero when the component $\vec{W}$ has a zero magnitude: such is the case for the theoretical trajectory.

I - 1.2 - Detection of the position of the moving body.

During the approach and the downward phases the missile location system consists of a radar, placed near the landing site, which provides spherical coordinates of the moving body M. These coordinates (distance D, sight angle S, and lie angle G). are measured from an origin R of the plane parallel to $\vec{X}\ \vec{Y}$ through R and, in this plane, from direction $\vec{X}$: Distance D is an essentially positive quantity, sight angle S (of magnitude smaller than $\frac{\pi}{2}$ radians) is positive if M is above the reference plane passing through R, the lie angle is measured in the counter-clockwise direction of plane $\vec{X}\,\vec{Y}$.

Point R is determined in the principal trihedron by the vectorial relation

$$\vec{AR} = a\vec{X} + b\vec{Y} + c\vec{Z}$$

The principal coordinates of the moving body are then

$$X = a + D \cos S \cos G$$

$$Y = b + D \cos S \sin G$$

$$Z = c - D \sin S$$

and the secondary coordinates X' and Y' can be written

$$X' = X \cos \delta + Y \sin \delta$$

$$Y' = - X \sin \delta + Y \cos \delta$$

The diagram on page 88 shows the orientation of the moving body relative to the earth at point A.

I - 1.3 - <u>Trihedron at take-off point</u>.

At point 0, origin of the trajectory, we can define a trihedron similar to $\vec{X}\,\vec{Y}\,\vec{Z}$. But, because of the curvature of the earth's surface, the unit vector $\vec{Z}$ forms an angle ϵ with the downward vertical at 0. The magnitude of ϵ depends upon the relative positions of 0 and A: for distance OA of the order of 300 kilometers, this angle ϵ is approximately 2.7 degrees.

Point 0 is, from the definition of the principal trihedron, located in plane $\vec{Z}\,\vec{X}$ passing through A:

$$\vec{AO} = \xi\,\vec{X} + \zeta\,\vec{Z}$$

If OA is always roughly 300 km and if both ends of the trajectory are at about the same altitude with respect to sea level, the coordinates ξ and ζ have magnitudes of the order of 300 and 8 km respectively. For any given pair of points O and A, the elements ξ and ζ are completely determined: It is therefore possible to take them into account, if need be, in the design of the cruise guiding system.

I - 2 - Coordinate frame linked with missile -- angular references

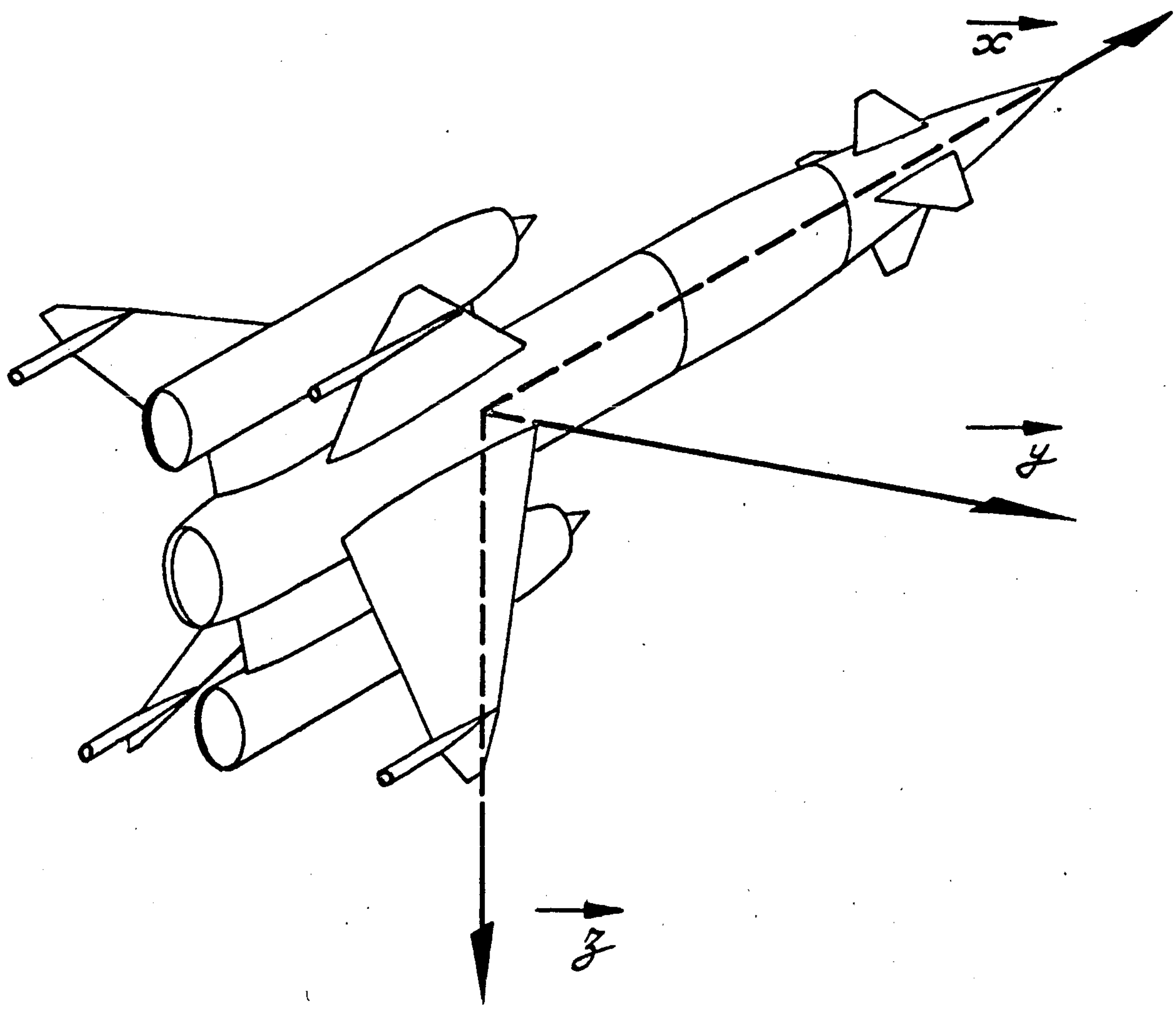
I - 2.1 - Definitions.

We consider three unit vectors with their origin at the center of gravity of the missile (diagram below):

$\vec{z}$ situated in the plane defined by the ramjet axes and perpendicular to the longitudinal axis of the missile (yawing axis),

$\vec{x}$ along the longitudinal axis in the direction of the nose (rolling axis),

$\vec{y}$ such that the frame $\vec{x}\,\vec{y}\,\vec{z}$ is orthogonal and right-hand (pitching axis).



I - 2.2 - Relation between "missile" trihedron and "principal ground" trihedron.

The orientation of the missile with respect to ground can be defined by means of a system of three angular quantities corresponding to the passage, through three successive rotations, from a trihedron of unit vectors $\vec{X}$, $\vec{Y}$, and $\vec{Z}$ to a trihedron of same origin and unit vectors $\vec{x}$, $\vec{y}$, and $\vec{z}$.

We shall use the system corresponding to the following series of rotations:

- the first one, of magnitude ν done about $\vec{Z}$, changes $\vec{X}\ \vec{Y}\ \vec{Z}$ into $\vec{X}_1\ \vec{Y}_1\ \vec{Z}$,

- the second one, of magnitude μ, done about $\vec{X}_1$, changes $\vec{X}_1\ \vec{Y}_1\ \vec{Z}$ into $\vec{X}_1\ \vec{y}\ Z_2$,

- finally the third one, of magnitude λ, done about $\vec{y}$, brings $\vec{X}_1\ \vec{y}\ \vec{Z}_2$ in coincidence with $\vec{x}\ \vec{y}\ \vec{z}$.

This reference system, which is the subject of the diagram on page 89, shows an indetermination only if the axis bearing $\vec{y}$ is perpendicular to plane $\vec{X}\ \vec{Y}$; there is no risk of the guiding system being planned to bring in the missile in this position.

The unit vectors of the two "principal ground" and "missile" trihedron are connected by a matrix relation which can be written, as a function of ν, μ and λ:

$$\begin{Bmatrix} \vec{x} \\ \vec{y} \\ \vec{z} \end{Bmatrix} = \begin{bmatrix} \begin{pmatrix} -\sin\nu\sin\mu\,\lambda \\ +\cos\nu\ \mathrm{cps}\,\lambda \end{pmatrix} & \begin{pmatrix} \cos\nu\sin\mu\sin\lambda \\ +\sin\nu\cos\lambda \end{pmatrix} & (-\cos\mu\sin\lambda) \\ (-\sin\nu\cos\mu) & (\cos\nu\cos\mu) & (\sin\mu) \\ \begin{pmatrix} \sin\nu\sin\mu\cos\lambda \\ +\cos\nu\sin\lambda \end{pmatrix} & \begin{pmatrix} -\cos\nu\sin\mu\cos\lambda \\ +\sin\nu\sin\lambda \end{pmatrix} & (\cos\mu\cos\lambda) \end{bmatrix} \begin{Bmatrix} \vec{x} \\ \vec{y} \\ \vec{z} \end{Bmatrix}$$

I - 2.3 - Angles between the "missile" axes and some of their projections.

We can formulate relationships involving the missile axes, the $\vec{X}\,\vec{Y}$ plane (horizontal plane at A) and the $\vec{Z}\,\vec{X}$ plane (plane containing the theoretical trajectory). The angles μ, θ, τ, ψ, σ and ρ which will be mentioned are shown on the diagram of page 90.

The angle between $\vec{y}$ and $\vec{X}\,\vec{Y}$ is no other than μ.

The angle θ between $\vec{x}$ and $\vec{X}\,\vec{Y}$ is no other than λ when μ is zero.

The angle τ between $\vec{z}$ and $\vec{Z}\,\vec{X}$ has a sine defined by

$$\sin \tau = -\cos \nu \sin \mu \cos \lambda + \sin \nu \sin \lambda$$

Let $\vec{x}'$ be the vector (not unit) orthogonal projection of $\vec{x}$ on $\vec{X}\,\vec{Y}$; when μ is zero this vector is shown by the intersection of plane $\vec{x}\,\vec{z}$ and plane $\vec{X}\,\vec{Y}$, and the angle ψ which it makes with $\vec{X}$ is equal to ν.

Furthermore let x'' be the vector (not unit) orthogonal projection of $\vec{x}$ on $\vec{Z}\,\vec{X}$ and σ the angle between $\vec{x}$ and $\vec{x}''$; when τ is zero we can verify the relation:

$$\cos \sigma = \cos \nu \cos \mu$$

Furthermore, and still when τ is zero, the angle σ defined above and the angle ρ between $\vec{x}''$ and $\vec{X}$ are simply related to λ by the equation $\operatorname{tg} \rho \cos \sigma = \operatorname{tg} \lambda$.

These various remarks will be used in the definition of the guiding laws. Let us mention at this point that we plan to keep the missile in the situation where τ is zero (yaw axis z is parallel to the plane containing the theoretical trajectory) during the cruise and the approach; on the other hand, during the descent, we shall attain a position with zero μ (pitching axis y is horizontal).

I - 3 - DETECTION OF THE ORIENTATION OF THE MISSILE

I - 3.1 - Gyroscopes aboard.

Two gyroscopes with two degrees of freedom are planned in order to measure, on board, missile attitude with respect to ground. The evolutions which the moving body will have to perform determine the arrangement of these gyroscopes.

Consideration of the theoretical trajectory shows that the angle between the longitudinal axis of the missile and $\vec{X}$ has an important variation corresponding to a rotation about $\vec{Y}$.

We shall see also that, during the descent, the plane $\vec{z}\,\vec{x}$ must contain the horizontal component $\vec{W}$ of the wind at A when $\vec{x}$ is not very far from the vertical (30 degrees at most): This situation corresponds roughly to a rotation (which can be important) about the vertical at A or about the rolling axis.

It follows from these two remarks that, in order to prevent the internal gimbal of the gyroscope from striking, we must have:

- either the rotor axis vertical and the axis of the external gimbal always roughly parallel to $\vec{Y}$,

- either the rotor axis parallel to $\vec{Y}$ and the axis of the external gimbal coinciding with the longitudinal axis of the missile.

Consider then a gyroscope, labelled number 1, which has (before release) its external gimbal parallel to the rolling axis, its internal gimbal parallel to the yawing axis and the rotor axis parallel to the pitching axis of the missile.

A gyroscope, labelled number 2, is mounted (before release) with its external gimbal parallel to the pitching axis, its internal gimbal parallel to the yawing axis, and its rotor axis parallel to the rolling axis of the missile.

I - 3.2 - Detections of the gyroscopes.

In starting position on its launching pad at 0, the missile has its longitudinal axis parallel to the vertical at A (and directed upward: $\vec{x}_0 = -\vec{Z}$) and its plane of pitch contains A ($\vec{z}_0 = \vec{X}$). The gyroscopes are released in the position so defined. When, after release of the gyroscopes and launching of the missile, the latter has any orientation with respect to the principal ground trihedron, we call $(-\alpha_1)$ the angle by which the external frame of gyroscope number 1 has rotated with respect to its locked position and $(-\beta_1)$ the angle of rotation of the internal frame with respect to the external frame of the same gyroscope; the quantities $(-\alpha_2)$ and $(-\beta_2)$ measure the same angles for gyroscope number 2.

The relations shown below connect α_1, β_1, α_2 and β_2 to ν, μ, and λ:

(1) $\sin \beta_1 = \cos \nu \sin \mu \sin \lambda + \sin \nu \cos \lambda$

(2) $\sin \alpha_1 \cos \beta_1 = \cos \nu \sin \mu \cos \lambda - \sin \nu \sin \lambda$

(3) $\cos \alpha_1 \cos \beta_1 = \cos \nu \cos \mu$

(4) $\sin \beta_2 = \sin \mu$

(5) $\sin \alpha_2 \cos \beta_2 = - \cos \mu \cos \lambda$

(6) $\cos \alpha_2 \cos \beta_2 = \cos \mu \sin \lambda$

Comparing relation (2) with the expression for $\sin \tau$ given at I - 2.3, we notice that α_1 and τ become zero together (provided β_1 does not reach $\frac{\pi}{2}$ radians, which is necessary for gyroscope number 1 to function).

But, if α_1 is zero, relation (3) compared with the expression for $\cos \sigma$ given in paragraph I - 2.3 when τ is zero, shows that β_1 and σ are equal when the axis $\vec{z}$ is parallel to the plane $\vec{Z}\,\vec{X}$.

Furthermore, relation (4) on one hand and relations (5) and (6) on the other hand, express that β_2 and μ are equal while α_2 is equal to $\lambda + \frac{\pi}{2}$.

Finally, if μ is zero, the rotation α_1 of the external frame of gyroscope number I is defined by $\operatorname{tg} \alpha_1 = - \operatorname{tg} \nu \sin \lambda$.

From now on we shall call e_1 and i_1 respectively, the information given by sensors placed at the external and internal gimbals of gyroscope number 1; likewise, e_2 and i_2 will represent the corresponding quantities relating to gyroscope number 2.

It follows from above that the two proposed gyroscopes can provide:

- λ from e_2 and μ from i_2 no matter what the orientation of the missile can be (e_2 also gives an approximation of ρ when τ is zero and σ near zero).
- σ from i_1 when τ is zero.
- a quantity e_1, which becomes zero with τ, and which constitutes, furthermore, an approximation for ν when μ is zero and λ is near $\frac{\pi}{2}$ radians.

As far as the approximation for ρ constituted by e_2 when the yawing axis $\vec{z}$ is parallel to the plane of the theoretical trajectory is concerned, it depends upon the value of σ. The diagram on page 91 gives the value of the difference (ρ - e_2) for various angles σ, as a function of ρ.

Likewise, the diagram on page 92 shows, for various values of λ and as a function of ν, the difference between the angle ν and the information e_1 when the pitching axis $\vec{y}$ is horizontal (in the neighborhood of A).

I - 3.3 - <u>Accuracy of the equipment.</u>

The orientation necessities which have been defined (yawing axis of the missile parallel to the plane of the theoretical trajectory or pitching axis horizontal) can be roughly within a few degrees without affecting the quality of the guidance. We have already shown that a tolerance of about twenty degrees on the orientation of the longitudinal axis of the missile was acceptable for the opening of the rotor.

Guidance is effected by using certain angles between the "missile" axes and the planes of the "principal ground" trihedron; sufficient accuracy is obtained with approximations of these angles. This is why we do not plan to use central directional control but simply two gyroscopes with their casings fixed to the missile. Moreover, since we are satisfied with values within a few degrees, we can use gyroscopes of rather common type.

The flight time of the missile (for the distance of 300 kilometers) is of the order of 11 minutes. During this time, the accelerations undergone by the equipment are small—except during certain periods of time when, in any case, the accelerations still remain smaller than 10 times the acceleration of gravity. By choosing gyroscopes whose drift, under the action of gravity, is of the order of 0.2 degrees per minute, we get errors of about 3 degrees on the orientation of the missile at the end of the flight.

Moreover, the motion of the earth corresponds as a first approximation to a rotation, with respect to space, of the principal trihedron defined at the landing point A. This rotation is about 3 degrees during the time of flight. It would be possible to add to the approach and descent computer, a correcting charge which would take into consideration the motion of the earth during the average time of a flight: such a stage would improve the quality of the guidance.

Considering the above, it is not necessary to seek an especially accurate device for the measurement of the gyroscopes deviations. We are simply planning detections with potentiometers whose accuracy can be of the order of one degree.

Reference Frames Linked with the Ground in the Vicinity of A

Diagram drawn with:

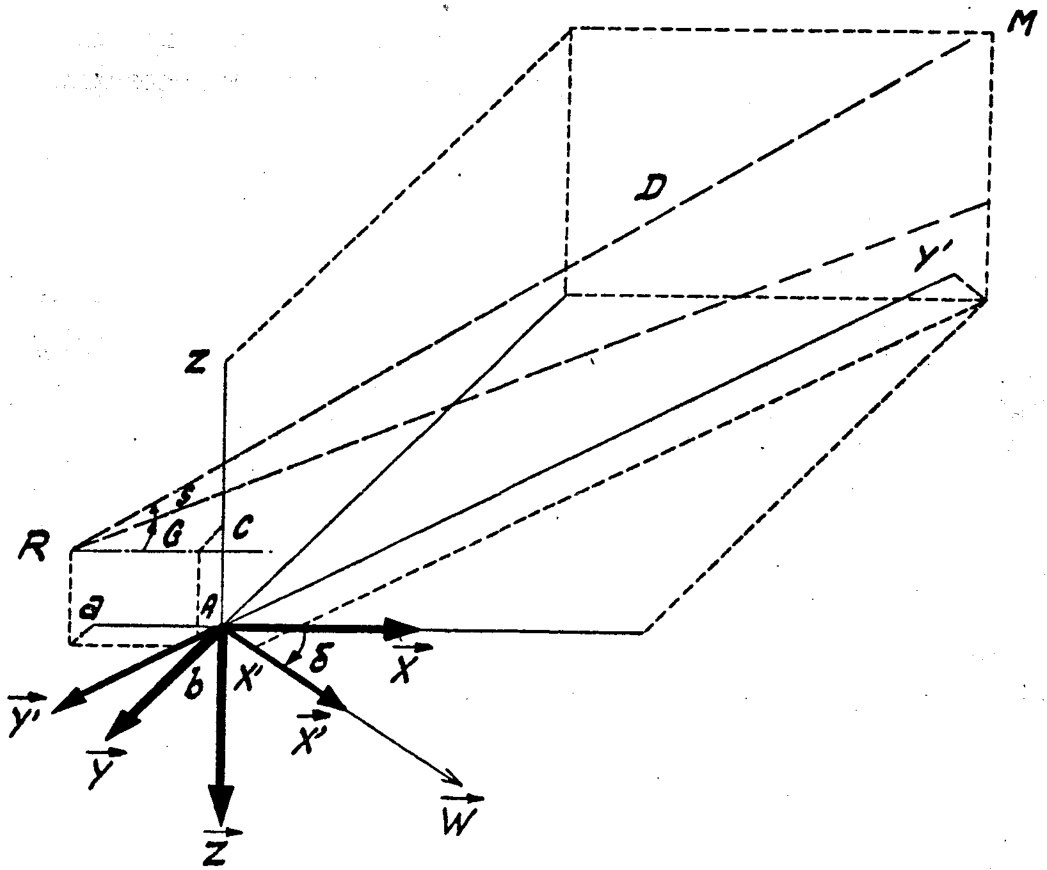
a - negative b - positive c - negative

D - positive s - positive G - negative

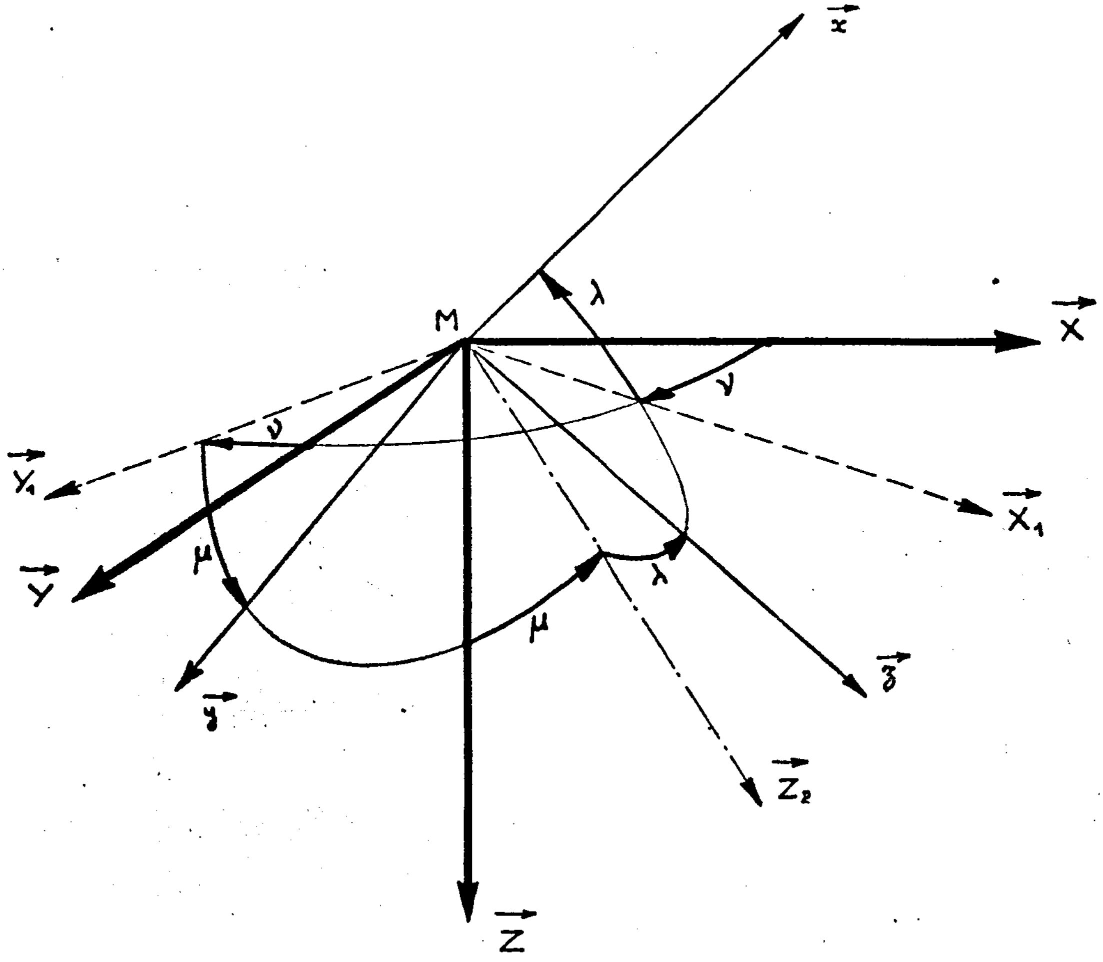
X - positive y - negative z - negative

δ - positive

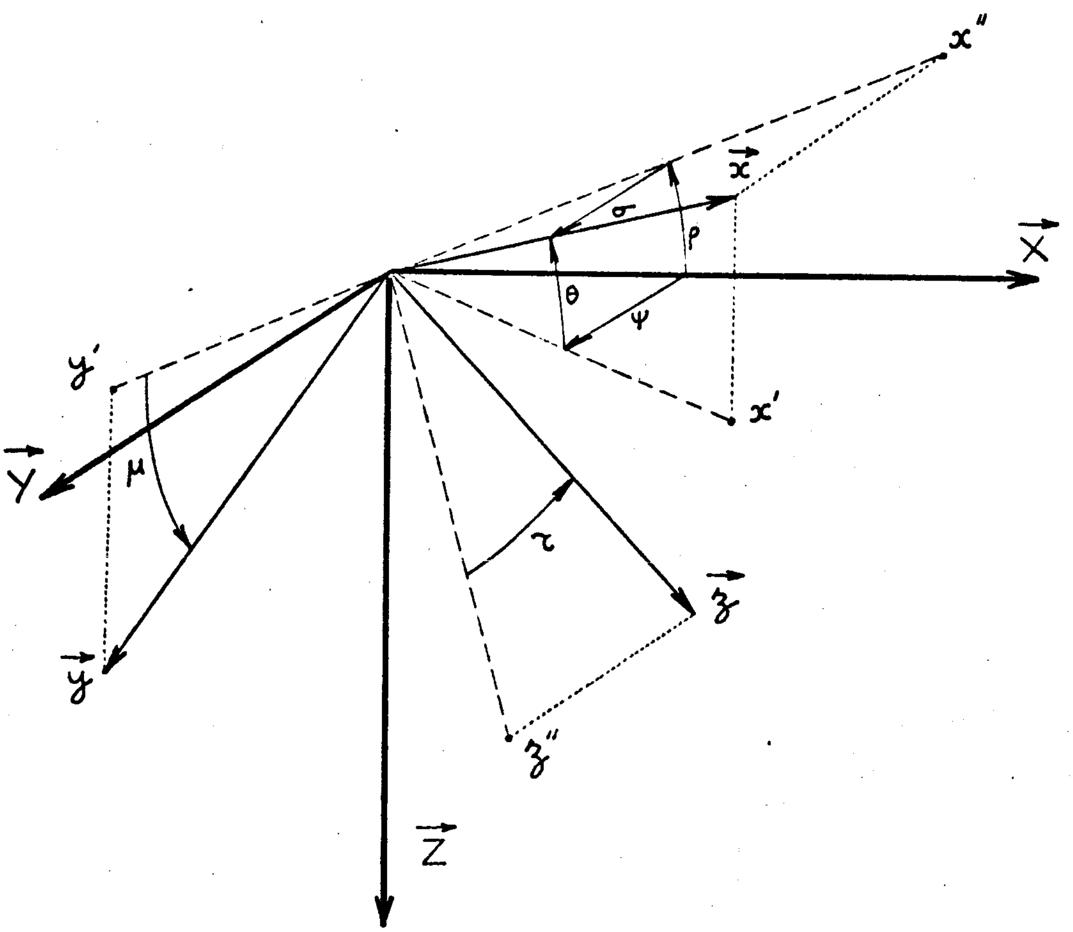
X' - positive Y' - negative



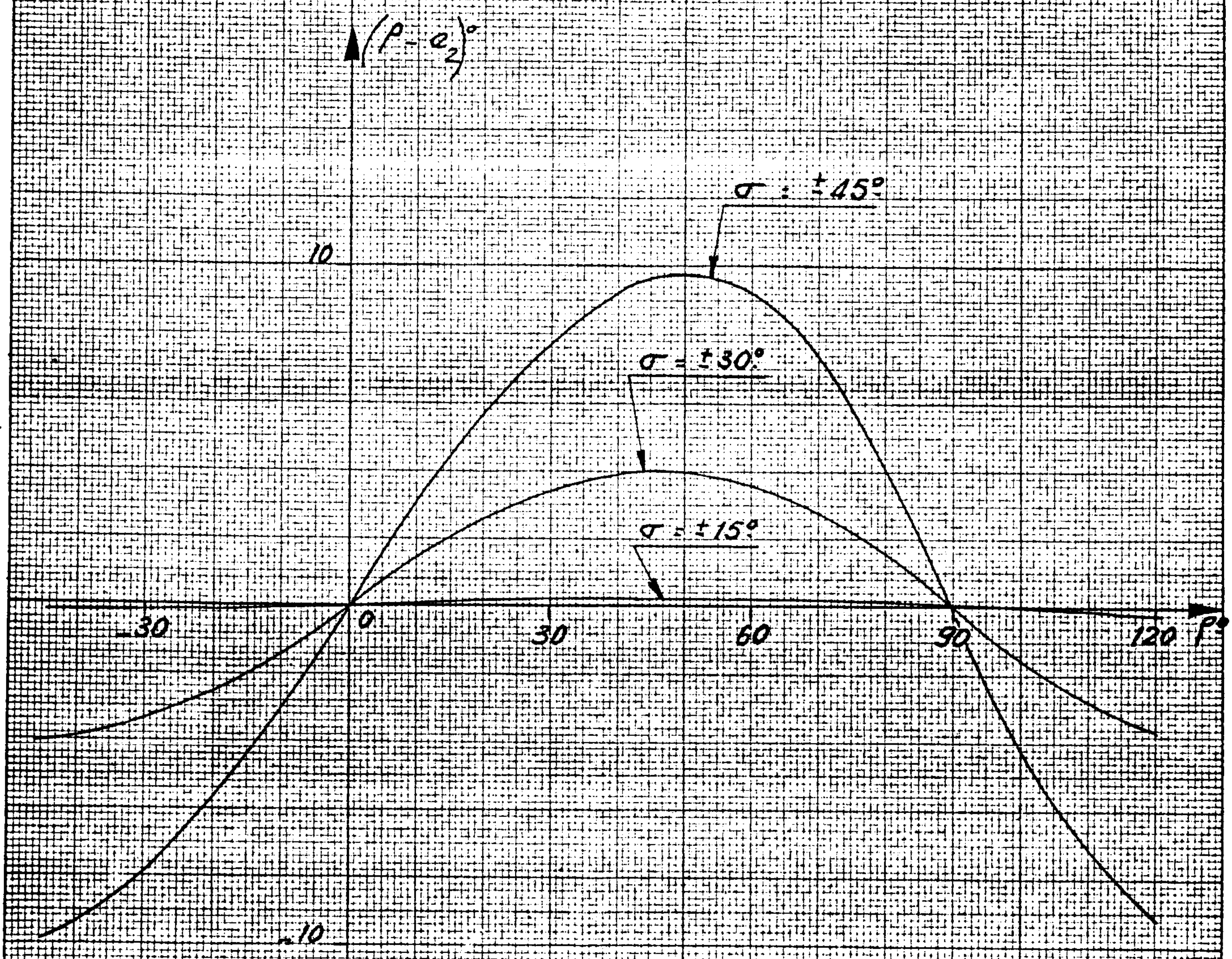
Angular References

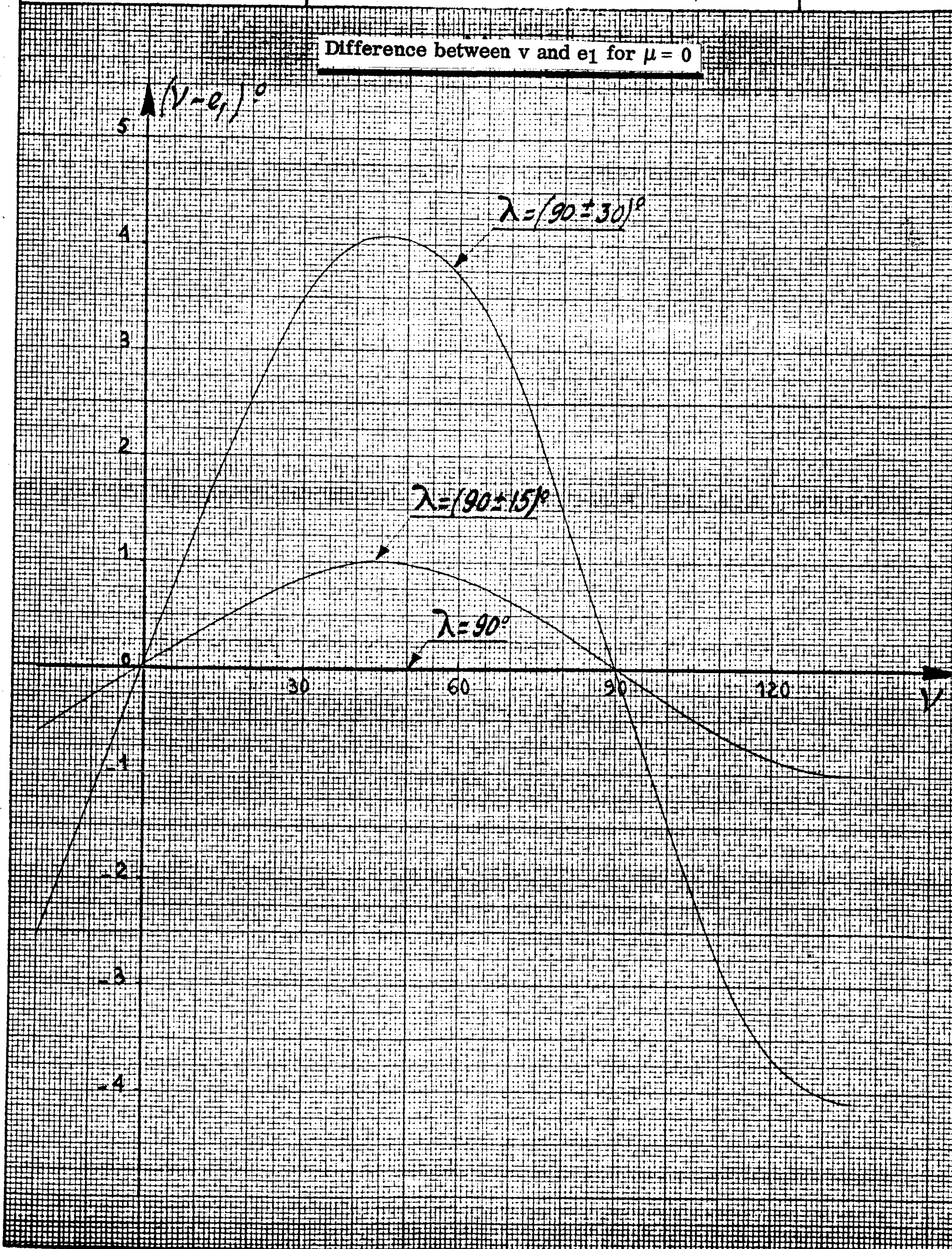


Angles of the missile axes with planes $\vec{x}\,\vec{y}$ and $\vec{Z}\,\vec{X}$



Difference between p and e_2 for $\tau = 0$





Difference between v and e_1 for $\mu = 0$
$(\nu - e_1)^o$
5
4
3
2
1
0
-1
-2
-3
-4
30
60
90
120
ν
$\lambda = (90 \pm 30)^o$
$\lambda = (90 \pm 15)^o$
$\lambda = 90^o$

APPENDIX II

APPROACH AND DESCENT COMPUTER

II - 4.4 - Determination of θ — program P1

II - 4.5 - Program for the tilting maneuver — program P2

II - 4.6 - Vertical maneuver program — Program P3

II - 4.7 - Sequence of operations performed by the computer during the approach and descent maneuvers.

II - 1 - GENERALITIES

The principles of the guidance of the missile during the approach phase and the "opened rotor" phase, are developed in chapter 5.

The guidance laws—capable of providing the planned maneuvers—remain to be defined. The kinetics of the motion are controlled from the ground where the information needed to establish the missile trajectory and the orientation of missile axes with respect to the reference system are calculated by the computer.

In fact, the statement of the guidance laws is tied to the computer programming. It is interesting to treat both problems simultaneously even if it is only to avoid too much complexity in the programming of the computer. We choose among the various statements of the laws of navigation those which best lend themselves to analog techniques.

All quantities worked-up in the computer are related to the reference trihedron $\vec{X}\ \vec{Y}\ \vec{Z}$, with origin A. Radar pinpoints the missile. Therefore, a first function of the computer will be to change the spherical radar coordinates D, S, G into cartesian coordinates X, Y, Z in the reference system fixed to the radar, parallel to the reference trihedron and of origin R; then into a translation defined by the segment $\overline{RA}$; finally giving the coordinates X, Y, Z of the missile in the reference system.

The study of the guidance laws and of the functions of the computer is divided into two parts: approach phase, and the "opened rotor" descent phase.

In a third part we present the operation of the computer.

II - 2 - APPROACH PHASE

II - 2.1 - Roll and Yaw Control.

In the course of this guiding phase, we have the following goals:

- to orient the axes of the missile in such a way that the command for vertical maneuver only affects pitch axis.

- to maintain the trajectory of the missile in the plane of guidance $\vec{X}\ \vec{Z}$.

Therefore, the first of these conditions implies a missile direction such that its axis $\vec{y}$ is perpendicular to the plane of guidance, that is u and ν small. The gyroscope detections e_1 and i_1, respectively, introduced in the rolling control system and in the yaw control system are suitable for effecting this direction.

The second condition can be readily satisfied by introducing, in the yaw control system, a term depending upon the coordinate Y. When the missile is close enough to the target, this term is cut out. This switching can be made at time t_4 defined at II - 2.2.

Thus the adopted guiding terms are:

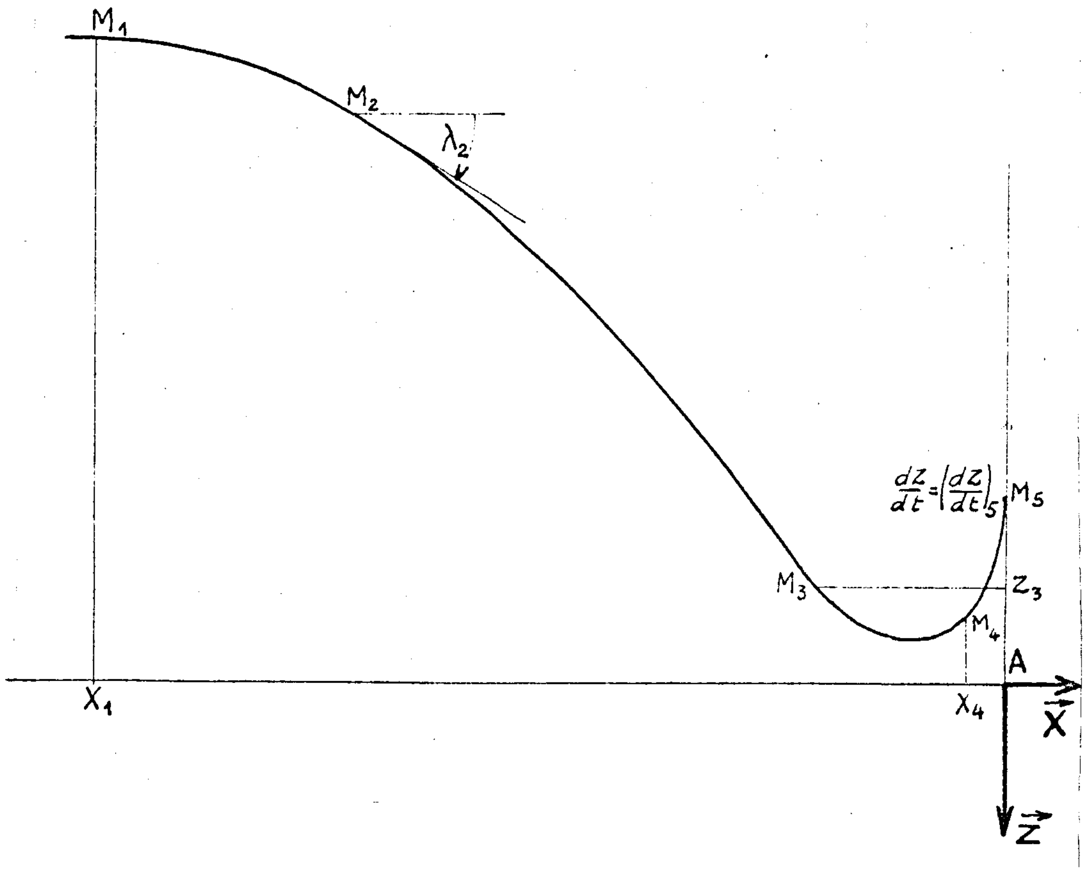
Roll - $K_o\ e_1$

Yaw - $(K_1\ Y + K_2\ i_1)$ $\quad t < t_4$

$- K_2\ i_1$ $\quad t > t_4$

II - 2.2 - Pitch control.

Missile commands are introduced in the pitching system. In its response, the missile dives and then pulls-up. On the diagram on page 97 are shown the specific points on the trajectory where modifications to the guiding laws are brought in; their definitions are given in the statements below where perfect roll and yaw response is assumed.



II—2. 21—Diving.

Point M_1 (X_1, Z_1), reached at line t_1 marks the end of the cruise; at this moment a signal from the ground by the guiding station triggers contacts aboard the missile which then performs the diving operations.

In the first sequence, diving is performed at the missile's design limit. The command is obtained by introducing in the pitching system a tension available aboard and suitable to bring the controls to the limits of their travel. The maneuver, controlled through limitations of load factors or incidence factors, lasts until time t_2, bringing the rocket to point

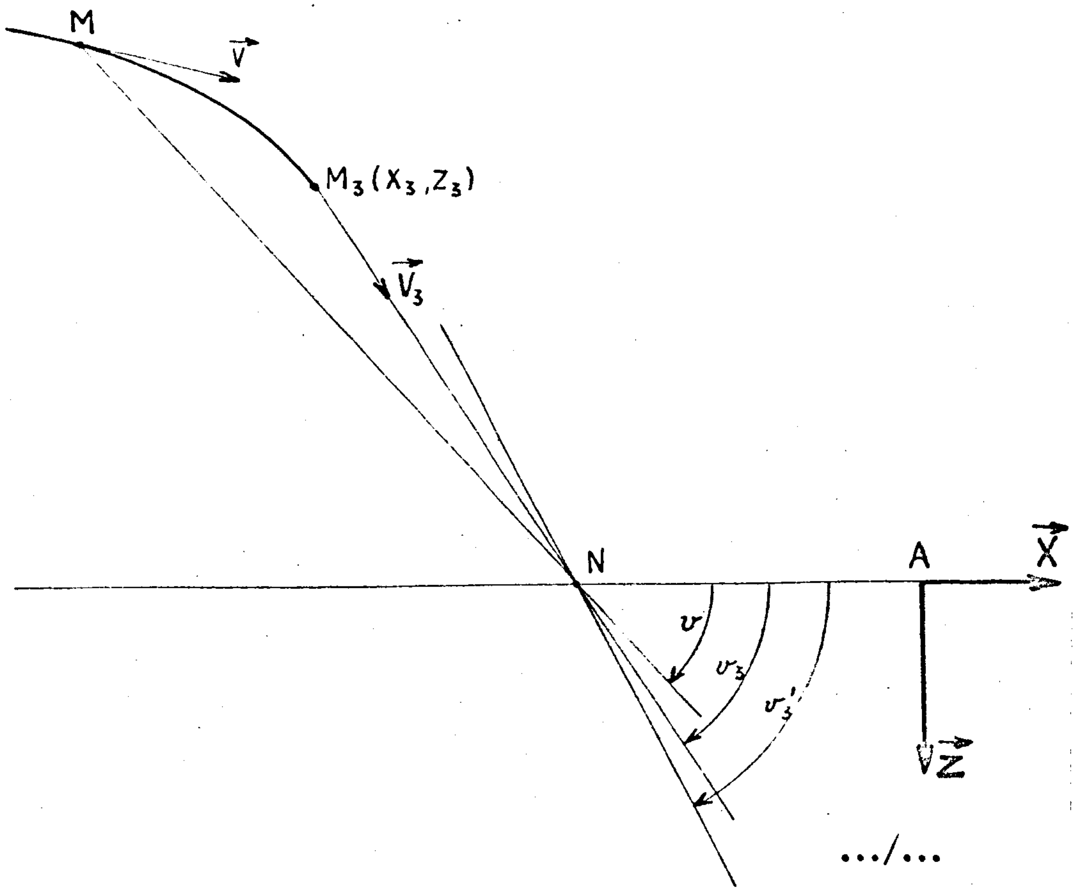
M_2 (X_2, Z_2), where the trim angle reaches the value λ_2 (- 45°, for instance).

At time t_2, a switch on board functions to change the "maximum performance" command to a ground command. This ground command was transmitted to the missile after t_1 and introduced into the pitch control system. This subsequent "controlled" dive maneuver should bring the missile to point M_3 (X_3, Z_3) in such a manner that at this point:

- The trajectory angle is at an angle $v_3 = (\vec{X}, \vec{V}_3)$: $\vec{V}_3$ = velocity vector at point M_3,

- The trim angle of the missile is equal to a given value λ_3.

These angles will be about the same and their absolute values will be close to λ_2.



.../...

This double aim can be reached simply, on the one hand, by matching the quantity v_3 with the angle $v = (\overrightarrow{MN}, X)$ [M: any point on the trajectory, N = intersection of the axis $\overrightarrow{X}$ with the line bearing the velocity vector $\overrightarrow{V_3}$ (see II—4.2 for the justification of this choice)]; and, on the other hand, by comparing the gyroscopic detection e_2 to the quantity λ_3.

In fact, to prevent the actual trajectories from showing an important "drag" due to gravity, a constant term, which is a function of the mass of the missile, must be added to the expression for guidance control. It is enough, in order to take the foregoing into account, to determine a value v'_3 larger than v_3.

Thus, during the diving phase, the adopted guiding laws are:

$$t_1 < t < t_2 \qquad \begin{cases} t_1 = t\ (X = X_1) \\ t_2 = t\ (\lambda = \lambda_2) \end{cases}$$

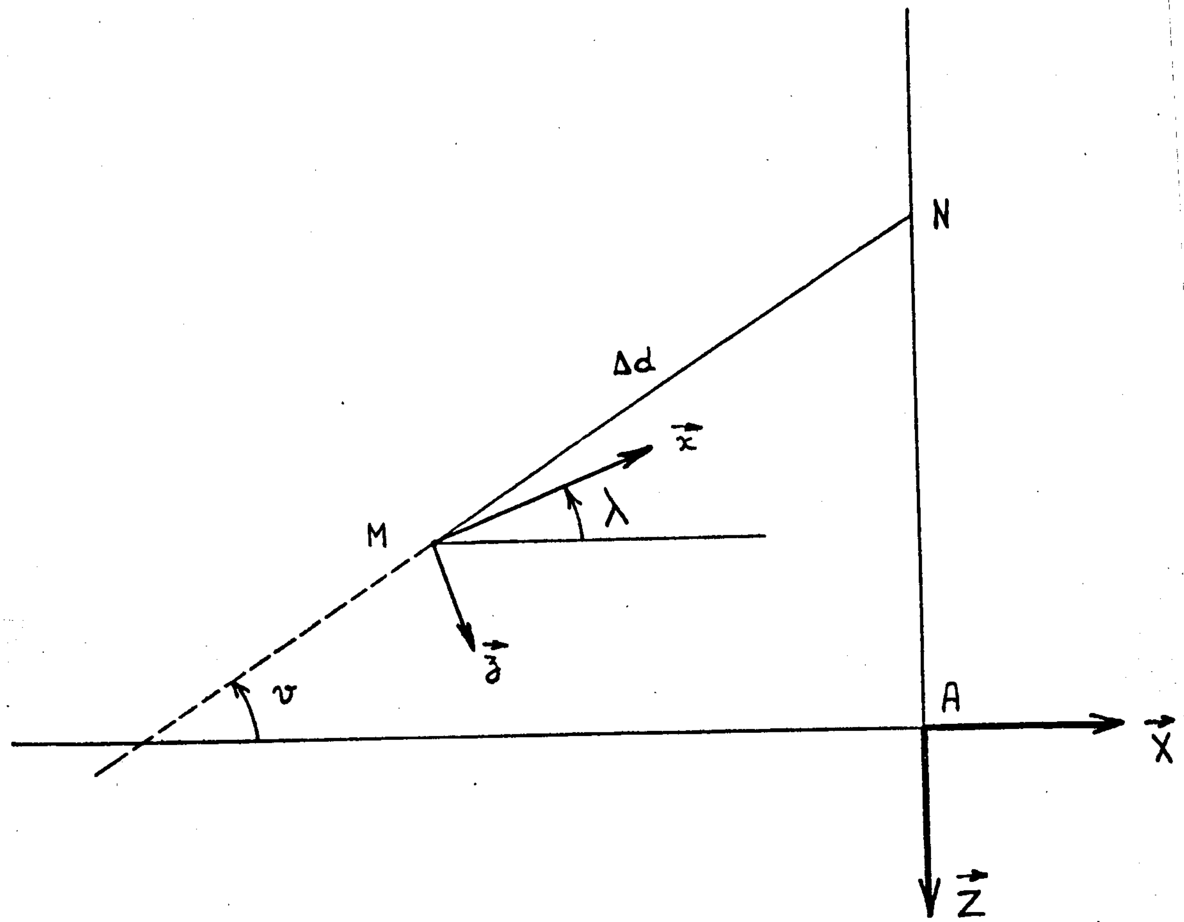
maximum brakage

$$t_2 < t < t_3 \quad , \quad t_3 = t\ (Z = Z_3)$$

$$K_3\ (v'_3 - v) + K_4\ (\lambda_3 - e_2)$$

II—2.22—<u>Pull-up</u>.

The diving stops as soon as $Z = Z_3$, at time t_3. At this time, the pull-up maneuver is triggered by ordering the missile, through a sequential command, to reach a point N on the Z axis.



N is fixed and its ordinate keeps a constant value H as long as the abcissa X of the missile does not become equal to a value X_4. From time t_4, when M passes through M_4 (X_4, Z_4), N moves along the axis $\vec{Z}$ in such a manner that its ordinate varies as the inverse of X and according to the expression $\frac{HX_4}{X}$.

Let $v = (\vec{MN}, \vec{X})$; the expression of the pitch control term during the whole pull-up is:

$t > t_3$

$$K_4 \; (v - e_2)$$

The advantage is that no switching is required aboard the missile when going from a dive to a pull-up. Only the ground order is modified: it becomes $K_4 v$ instead of $K_3 (v_3 - v) + K_4 (\lambda_3)$.

II—3—"Opening rotor" descent phase

II—3. 1—Introduction.

During the last moments of the approach, more precisely when $|X|$ reaches a value close to $|X_4|$, the climbing rate $\left|\frac{dZ}{dt}\right|$ is checked against a given value $\left|\left(\frac{dZ}{dt}\right)_5\right|$ which, once reached, marks the end of the approach phase.

At that instant t_5, a signal transmitted from the ground orders both the opening of the rotor and switchings aboard the missile which will permit the guiding in this new dimension.

In addition to the guidance terms meant to determine the orientation of the "missile" trihedron and to correct the heading errors with respect to the objective as in the approach phase, orders controlling the descent are transmitted to the missile.

The heading errors with respect to the target A on the horizontal plane are given in a system of axes A, $\vec{X'}$, $\vec{Y'}$ defined with respect to A, $\vec{X}$, $\vec{Y}$ by a rotation δ such that the axis $\vec{X'}$ becomes parallel to the direction of the wind.

Recalling the statement on the guidance principles during the "opened rotor" descent, developed at 5-4, three main phases can be distinguished:

- intermediate phase consisting of putting the missile into free-rotation descent, that is, "autorotative" descent.

- Free rotation descent,

- pull-up: this phase is triggered by a tipping maneuver preceding the pseudo-stabilized phase leading to the landing of the missile.

II—3. 2—Roll and yaw control.

All along the trajectory, the plane $\vec{x}\ \vec{z}$ of the "missile" trihedron is maintained in coincidence with the plane $\vec{X'}\ \vec{Z'}$ of the target trihedron.

As soon as the rotor opens, the missile rotates through an angle δ about the $\vec{x}$-axis. The command, and then the keeping of the plane $\vec{x}\ \vec{z}$ parallel to plane $\vec{X'}\ \vec{Z}$ are achieved by setting of $\nu = \delta$ and $\mu = 0$ (since λ is close to 90 degrees). The gyroscope detections e_1 and i_1 are suitable for this setting.

By introducing in the yaw system a term proportional to Y', we bring the planes $\vec{x}\,\vec{y}$ and $\vec{X'}\,\vec{Z}$ into coincidence.

Thus, the guidance terms adopted are:

Roll: $K_7\ (\delta - e_1)$

Yaw: $- K_5\ Y' - K'_5\ i_2$

The adaptation coefficient K_5 is not constant during the motion.

During the intermediate phase consisting of putting the missile into free-rotation descent, K_5 increases from the value zero to the constant value used during the descent as is justified below (II—3.31).

II—3.3—Pitch control.

II—3.31—Intermediate phase consisting of putting the missile into free-rotation descent.

This phase is meant to place the missile into a flight position suitable to permit its guidance during the free-rotation descent:

- The rotor is opened and started.

- The pitch-plane orientation is set.

- Moreover, the missile has to tip its longitudinal axis by an angle θ, depending on velocity of the wind W. To achieve this, θ is checked against e_2 in the pitch control system. In order to take into account the reduced efficiency of the rotor during the period while it is getting up to speed, the guidance terms X' and Y' are introduced progressively.

The pitch-control law adopted during this phase has the expression:

$$- K_5\ X' + K_6\ (\theta - e_2)$$

K_5 varies from zero to the constant value used during the free-rotation descent.

II—3.32—Free-rotation descent.

Assume a perfect response of the axes of the missile around imposed directions. What remains is for the missile to correct the errors X' and Y' during the free-rotation phase.

Nothing is changed in the pitch guidance law except that K_5 is kept constant.

Guidance law:

$$- K_5 X' + K_6 (\theta - e_2)$$

II—3.33—Pull-up phase.

The descent velocity of the missile during the preceding phase is too great to permit landing; a pull-up is necessary.

A signal transmitted from the ground as soon as, at time t_6, the missile reaches the altitude Z_6, triggers the tilting maneuver.

The tilting order, worked-up from the ground, is introduced in the pitch-control system by substituting itself to the guidance terms.

During the tilting maneuver, at time t_7, a signal also transmitted from the ground, triggers a change in the general pitch of the rotor blades and starts the motors, thereby bringing to the rotor the energy necessary for the performance of the pull-up.

The general pitch order is determined according to altitude and descent speed; the error terms, coming from a ground signal, correct the constant pull-up control contained in the missile.

At time t_8, the tilting maneuver is completed.

The missile then enters the so-called "pseudo-stabilized" phase where the pitch guidance as a function of X' and θ is resumed until the missile reaches the ground.

Thus, during this phase, the guidance laws are:

Pitch:

$t_6 < t < t_8$ $\quad$ $t_6 = t \ (Z = Z_6)$

tilting command, programmed as a function of time

$t > t_8$ $\quad$ $t_8 = t_6 + \Delta t_2$

$- K_5 X' + K_6 (\theta - e_2)$.

In general:

$t < t_7$ $\quad$ $t_7 = t_6 + \Delta t_1$ $\quad$ $(\Delta t_2 > \Delta t_1)$

no command

$t > t_7$

command programmed as a function of z and $\left|\frac{dZ}{dt}\right|$.

II — 4 — COMPUTER OPERATION

II — 4.1 — Conversion of coordinates.

The radar data are delivered by potentiometers, sine-cosine for position and bearing, linear for range. By means of three transformer amplifiers (Amplis 1, 2, and 3), we obtain directly the Cartesian coordinates $\mathcal{X}$, $\mathcal{Y}$, $\mathcal{Z}$ of the missile with respect to the radar trihedron. Coordinates X, Y, Z with respect to the principal trihedron are given by amplifiers 4, 5, 6, and 7 on the functional diagram on page 113.

II — 4.2 — Determination of angle v.

Amplifiers 9, 10, 11 and 12 give the components ΔX, ΔZ of the vector $\overrightarrow{MN}$.

Servomechanism II, whose error signal is picked up by amplifier 13, solves the equation

$$\frac{\Delta X}{\Delta d} \sin v - \frac{\Delta Z}{\Delta d} \cos v = 0$$

and furnishes angle v over a linear potentiometer. Data from the sine-cosine potentiometers of this servomechanism, moreover, give the data necessary for the derivation of $\Delta d = \Delta X \cos v + \Delta Z \sin v$; servomechanism III uses v for working out Δd.

The position of point N on plane $\vec{X}\ \vec{Z}$ depends on the flight phase of the missile during the approach.

<u>Dive</u>: Point N is not identical with M_3, so that v is never indeterminate; its coordinates are:

$$N \quad \left| \begin{array}{l} X'_3 = X_3 - Z_3 \operatorname{cotg} v_3 \\ Z'_3 = 0 \end{array} \right.$$

Nose up: Point N is on the axis $\vec{Z}$, but its height varies as a function of X.

$$|X| > |X_4| \qquad N \left| \begin{matrix} O \\ \\ H \end{matrix} \right. \text{, constant}$$

$$|X| < |X_4| \qquad N \left| \begin{matrix} O \\ \\ H \cdot \dfrac{X_4}{X} \end{matrix} \right.$$

Amplifier 8, whose gain is servoed to X through servo-mechanism I, constantly provides the $H \cdot X_4/X$ figure, which is compared to H by means of the polarized relay $R_p 4$.

A regulator limits the variation of $H \cdot X_4/X$ to 5,000 meters.

The coordinates of point N are matched against the coordinates of the projection of the missile's position on the $\vec{Z}$ $\vec{X}$ plane over amplifiers 9 and 11 via time-constant integrator nets such as to prevent the servomechanism II, furnishing v, from becoming deservoed at the instant of the "dive-nose up" switch. The maximum value of this time constant is determined as a function of the duration of the completed flight, which starts the missile's nose up.

II — 4.3 — Rotation of the trihedron as a function of wind direction.

Angle δ is set manually. Its value is taken onto a linear potentiometer coupled to two sine-cosine potentiometers used for rotation of the principal trihedron during the "rotor spread" stage of the descent.

Amplifiers 6, 7 and 11, 12, respectively furnishing Y and X with the two signs, feed the two sine-cosine potentiometers, and their data summed on amplifiers 15 and 16 give coordinates X' and Y' of the missile with respect to the new system of axes.

$$X' = X \cos \delta + \sin \delta$$

$$Y' = -X \sin \delta + Y \cos \delta$$

Amplifiers 15 and 16 in fact furnish quantities proportional to X' and Y', more exactly, K_5 X' and K_5 Y'. By means of a program motor M, set in motion by signal S_2, K_5 is made variable, varying from zero to the constant value fitting the guidance used during the autorotation phase of descent.

II — 4.4 — Determination of θ — Program P1.

The inclination of the lateral axis of the missile as a function of wind velocity W is obtained by making an angle θ, as opposed to λ in the missile, correspond to W.

$\theta = f\ (W)$ is obtained by means of a program P1 which furnishes $K_6\theta$

W is set as an input to amplifier 21.

II — 4.5 — Command for the dipping maneuver - Program P2.

The polarized relay Rp 6 starts program P2 when missile altitude Z reaches the value Z_6.

The functions of program P2 are:

a) transmit signal S_3 (instant t_6),

b) open the pitch guidance loop by energizing relay R6, near amplifier 19, and insert the dip command which is derived as a function of time,

c) start program P3 (instant t_7),

d) cut signal S_3 (instant t_8); relay R6 settles back, allowing the pitch guidance program to resume. P_3 is then out of the circuit.

II — 4.6 — Vertical maneuver program — Program P3.

Program P3 works out error terms as a function of altitude Z and rate of descent $\left|\frac{dZ}{dt}\right|$. These terms correct the constant load factor-order stored in the missile.

Program P3 is started at time t_7 by signal S_4 put out by P2.

II — 4.7 — Sequence of operations performed by the computer during the approach and descent maneuvers.

The data needed to work out the guidance commands are continuously determined no matter what the maneuver of the missile; they are adapted beforehand to the scales desired (amplifiers 15, 16, 17, 18, and 22). Switches select the figures needed at any particular moment; and they are passed on to the four continuous channels G_1, G_2, G_3, and G_4 of the remote control transmitter.

The computer also has the function of setting up the signals needed to command certain commutations to be performed aboard the missile. These signals are given to the four channels S_1, S_2, S_3, and S_4 of the remote control.

During an approach maneuver the computer performs the following functions:

The polarized relay Rp 1 compares coordinate X with a fixed value X_1. When $X = X_1$, signal S_1 goes out and, through relay R_1, permits transmission of the figures worked out by the computer to the continuous channels.

Angle v is matched agains quantity $v'_3 + K_4/K_3 . \lambda_3$ set on a potentiometer.

The quantities transmitted are:

$$t_1 \langle t \langle t_3$$

$$G_1 = K_3 \left[\left(v'_3 - v\right) + \frac{K4}{K3} \lambda_3 \right]$$

$$G_2 = K_1 Y$$

$$G_3 = G_4 = 0.$$

When Z reaches value Z_3, polarized relay Rp 3, through self-restoring relay R 31, directs the following communications:

a) relays R 3.2, R 3.3, and R 3.4: change in scale of distances, with the objective of reducing the systematic errors accruing to the computer in working with higher voltages,

b) relay R 3.5: relaying of point N,

c) relays R 3.6 and R 3.7: in the determination of G_1, v changes sign and quantity $v'_3 + \frac{K4}{K3} \lambda_3$ no longer appears in its expression.

The quantities transmitted are:

$t_3 < t < t_4$

$$G_1 = K_4 v$$
$$G_2 = K_1 Y$$
$$G_3 = G_4 = 0$$

The dive maneuver takes place, the missile is guided toward fixed point N(O, H) just at the moment when X reaches the value X_4, and at this instant, by action of polarized relay Rp 4, point N ceases to be fixed as previously stated in II — 2.22. Furthermore, for a value close to X_4, polarized relay Rp 2 commands relay R 2 which cancels command G_2.

The quantities transmitted are then:

$t_4 < t < t_5$

$$G_1 = K_4 v$$
$$G_2 = G_3 = G_4 = 0$$

Amplifier 14 furnishes climb rate $\frac{dZ}{dt}$. This derivative can be obtained with a satisfactory signal/noise ratio in the frequency band used by adopting a suitable filter.

Polarized relay Rp 5 compares $\frac{dZ}{dt}$ to $\left(\frac{dZ}{dt}\right)_5$ (fixed figure) and dips when equality is reached, transmitting signal S_2. The descent maneuver begins. Relay R_5 performs commutation of the data to be transmitted, which are:

$t_5 < t < t_6$

$$G_1 = K_5 \left(X' + \frac{K6}{K_5}\theta\right)$$

$$G_2 = K_5 \; Y'$$

$$G_3 = K_7 \, \delta$$

$$G_4 = 0$$

The missile, in autorotative descent, reaches altitude Z_6; at this moment, t_6, polarized relay Rp 6 starts program P2.

Program P2 carries out the following functions:

a) transmits signal S_3 to the missile,

b) commands relay R_6 which cuts the terms which are functions of X' and θ out of the circuit and permits transmission over channel G_1 of the dip command it works out,

c) starts program P3 (at instant t_7) which sends signal S_4 to the missile and the error terms of the command for general pitch over channel G_1.

The quantities transmitted are therefore:

$t_6 < t < t_7$

$$G_1 = \text{dip order}$$

$$G_2 = K_5 \; Y'$$

$$G_3 = K_7 \, \delta$$

$$G_4 = 0$$

$t_7 < t < t_8$

G_1 = dip order

$G_2 = K_5 \; Y'$

$G_3 = K_7 \; \delta$

G_4 = command for general pitch

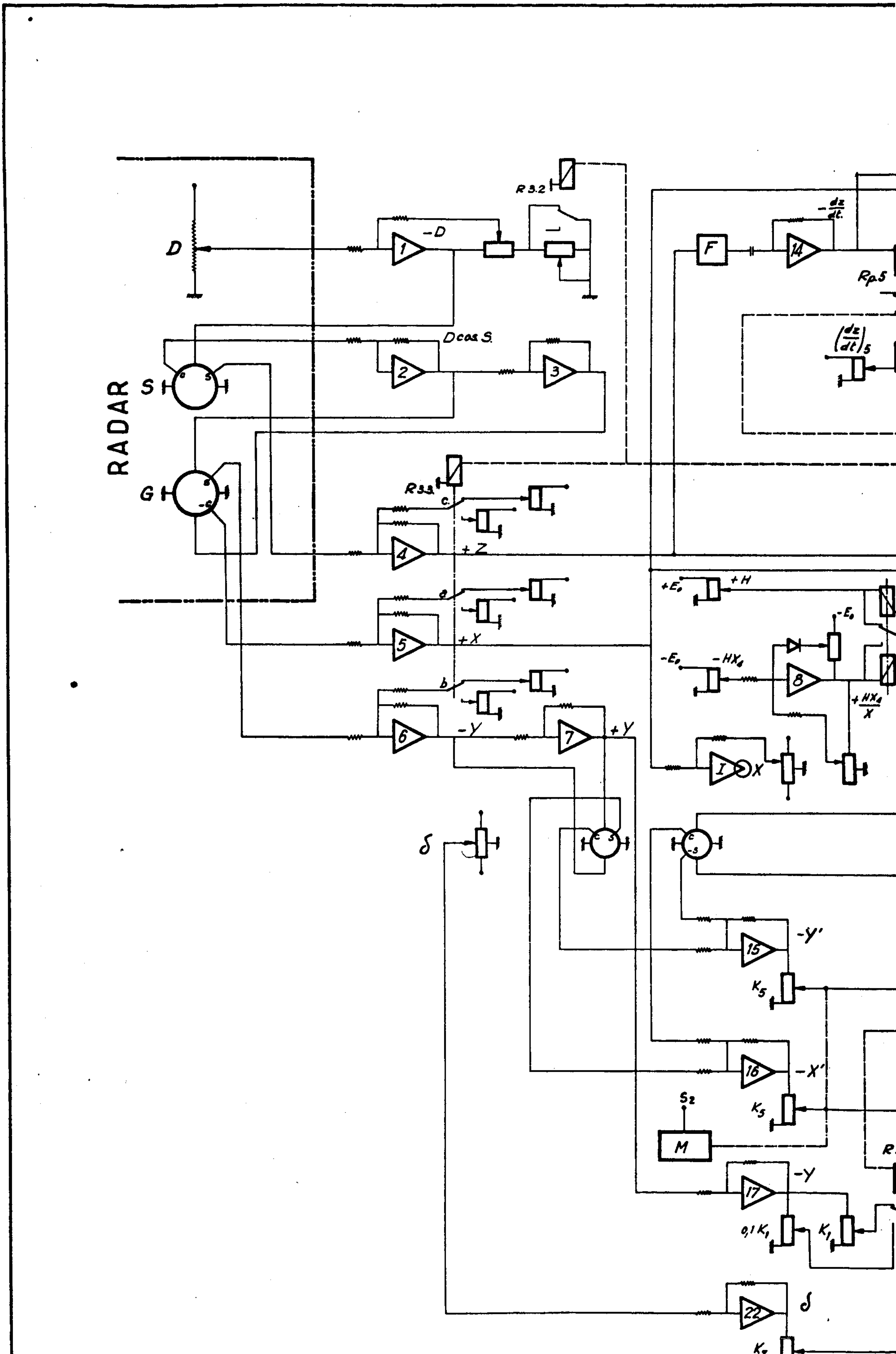
$t > t_8$, on the ground

$G_1 = K_5 \left(X' + \frac{K_6}{K_5} \theta \right)$

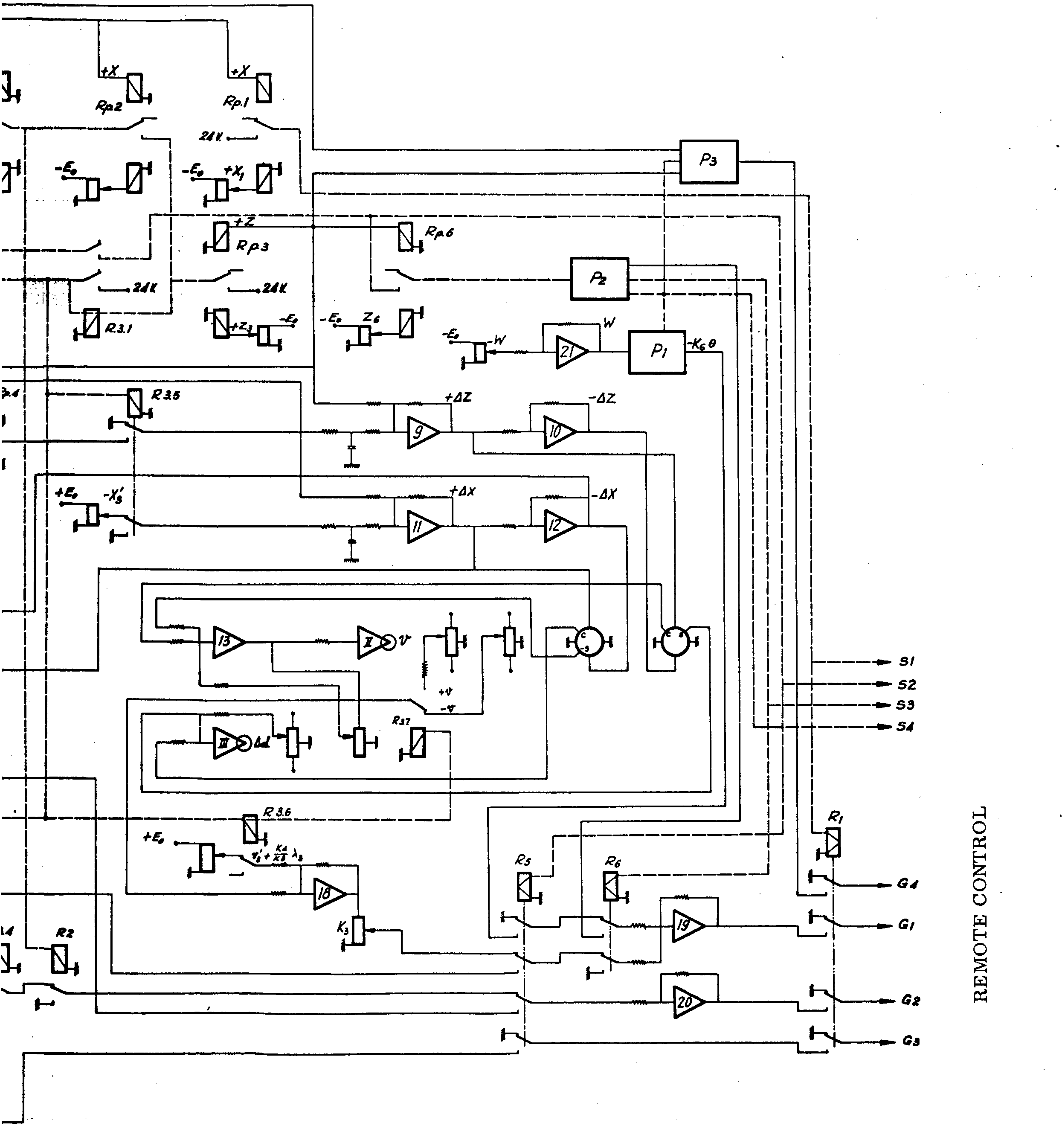
$G_2 = K_5 \; Y'$

$G_3 = K_7 \; \delta$

G_4 = command for general pitch.


RADAR
D
-D
R 3.2
D cos S
S
G
R 3.3
c
a
b
+Z
+X
-Y
+Y
δ
F
$-\frac{dz}{dt}$
Rp 5
$\left(\frac{dz}{dt}\right)_5$
$+E_0$
+H
$-E_0$
$-HX_4$
$+\frac{HX_4}{X}$
X
-Y'
-X'
K_5
S_2
M
-Y
$0,1K_1$
K_1
δ
K_7

113-1



+X
Rp.2
Rp.1
24V
P3
P2
P1
Rp.3
Rp.6
+Z
R3.1
R3.5
R3.6
R3.7
W
-K6θ
+ΔZ
-ΔZ
+ΔX
-ΔX
R5
R6
R1
R2
K3
S1
S2
S3
S4
G4
G1
G2
G3
REMOTE CONTROL

APPENDIX III

GUIDANCE SYSTEM

III - 1 - INTRODUCTION

This appendix gives a description of the missile-borne equipment that is necessary to control the maneuvers of the missile and to ensure its stability during various flight phases.

The overall design of the automatic pilot is intended to minimize the number of components used and at the same time to ensure the following three essential functions:

- Control (rolling - pitching - yawing) by means of fins (ascent, cruising, and approach phases),

- Control (pitching - yawing - vertical descent) by means of the rotor during descent with "unfolded rotor",

- Rolling control by means of fixed nozzles that are on-off devices: off for descent with unfolded rotor.

We shall describe successively the components of the pilot and its operating principle, which is illustrated by a block diagram on page 123.

III - 2 - AUTOMATIC PILOT COMPONENTS

The receiver of the remote control system delivers guidance and control signals worked out on the ground by the computer. The missile-borne equipment used for guidance is the following.

III - 2.1 - Detection units

Missile trim

Two free gyroscopes deliver the three reference angles for missile attitude control. Their adjustment on board has been described in Appendix I.

Angular velocities

Three gyrometers give the necessary damping terms around the guidance axes, rolling-pitching-yawing.

Load factors

Two on-board accelerometers are used to limit the load on the missile during rolling and yawing.

The maneuver commands during rolling and yawing are worked out by comparing trim data with the corresponding command data.

Command interruptions due to load could occur primarily during the cruising and approach phases when dynamic pressures are high. Besides, the commands will be subject to a saturation principle to limit their number at low dynamic pressures.

Angular detection of rolling about a zero line will be used during the cruising and approach phases. During descent with "unfolded rotor" all or none control by nozzles will exploit the difference between the angular reference for rolling and wind direction transmitted by remote control.

III - 2.2 - Power units

The large number of command units used impelled us, for weight reasons, to use hydraulic control.

A common hydraulic generator, comprising the pump, its driving motor, and supply tank, will supply two circuits successively:

- the circuit of the four command jacks for aerodynamic controls,

- the circuit of the three command jacks for the rotor control.

The nozzles used to control rolling during descent will be supplied with compressed air stored in the central tube. The amount of air necessary, which depends primarily on operating time and on the friction couple developed by the rotor, will be of the order of 12 kg.

III - 2.3 - Summation chains (loops)

The automatic pilot comprises 4 summation chains used in the following manner:

Before the rotor unfolds, chains (1) and (2) work out, respectively, the commands for positioning the pitch and yaw control fins. Chains (3) and (4) control the differential position of rolling control fins.

During descent with "unfolded rotor" chains (1), (2), and (3) work out the commands for activating the three rotor control jacks. Chain (4) works out the angular deviation due to rolling, i.e., the difference between gyroscope detection and wind direction transmitted by remote control. This difference, as well as the angular rolling speed, are utilized by a switching logic that controls the opening of the nozzles.

III - 2.4 - Switching devices

Besides the switching logic for rolling control by the nozzles, the main missile-borne switching units are activated by remote commands, all or none, triggering the following operations:

- dive command that initiates the approach phase,
- release of the sequence of rotor unfolding,
- command for transition maneuvers preceding landing.

Transmission of the various on-off control signals is triggered by the computer on the ground.

III - 2.5 - Programming motor

The sequence of the unfolding of the rotor and controls of the associated guidance chain will be irreversibly controlled by a programming motor aboard the missile. This motor is activated by remote control from the ground.

III - 2.6 - Adaptation units

The block diagram of the automatic pilot does not show the correction systems and gain control units that will have to be decided upon during subsequent adaptation studies. Let us note that the flight patterns at low speed preceding the unfolding of the rotor and autorotative descent during wind could require very elaborate adaptation units.

III - 3 - ACTIVATION OF THE AUTOMATIC PILOT

III - 3.1 - Ascent - cruising - approach

During these three phases the missile is guided by the fins that are controlled by chains (1) and (2) for pitching and yawing, and (3) and (4) for differential braking during rolling. The remote control command for rolling is zero and the pitching and yawing remote control channels deliver the respective guidance commands successively and continuously.

The guidance command for pitching during the ascent phase is programmed on the ground.

The guidance command for yawing is zero until the missile levels off.

The cruising flight is interrupted by a remote control command all or none (S_1) that stops the guidance command for pitching and replaces it by a maximum command for diving delivered by the automatic pilot. When the pitching angle detected on board reaches -45°, the command (S_1) is stopped and the guidance command for pitching is re-established (the switching device is not illustrated).

III - 3.2 - Sequence of rotor unfolding

This sequence is programmed as a function of time.

The programming motor is activated by remote command (S_2) when the missile reaches the desired vertical speed. We were guided by the following considerations when establishing the sequence of the operations. Firstly, a delay of several tens of seconds is necessary for the ports to open fully and for the ignition of the rotor propellant. This time interval can be used to stop the command that governs the rolling control fins and to activate rolling control by nozzles. We believe that this arrangement is desirable to ensure correct rolling control immediately after the appearance of the friction couple in the rotor hub.

After the powder propellant ignites, the time necessary to activate the rotor would be of the order of 1 sec. The operations for placing the rotor under control will be blocked during this time interval in order to maintain pitch-yaw control on the canard fins of the missile and to switch to control by rotor when it is active enough. Evidently, the delay must be as small as possible in all the switching operations of the automatic pilot system.

The succession of orders given by the programming motor during unfolding may be the following:

- command for opening the ports (at the very instant when signal S_2 is given, t = 0),
- stopping commands for rolling control at the input of summators (3) and (4),
- hydraulic locking of the rolling control jacks,
- establishment of the deviation due to rolling at the input of summator (4) and feeding the information into the switching logic,
- activation of the rolling control nozzles,
- ignition of rotor propellants (t = 0.5 sec).

Activation of the rotor promotes automatically:

The emergence of the telescopic element of the blade that is braked by a cable which is paid out by means of an irreversible gear at the rate of 3 m/sec.

The unlocking of the blade arm when centrifugal force exceeds a predetermined value.

The unfolding of the rotor that is braked by the paid-out cable.

Inactivation of the canard fins is initiated during unfolding and comprises the following operations:

- stoppage of commands for pitch-yaw control at the input of summators (1) and (2),

III - 3.5 - Pull-out

Signal S_3 cuts the gyroscopic reference at the input of summator (1) and at the same time the "successive" remote pitch-control channel delivers a programmed sawtooth signal that alters the trim. Annulment of the signal (S_3) after a determined time re-establishes the gyroscopic reference, and at the same time the tilt signal is annulled on the "successive" remote control channels and replaced by a trim program that corresponds to the pseudo-stabilized pull-out phase.

Vertical maneuver during pull-out is achieved through tracking units on the ground. A "successive" remote-control channel (pitch control, in general) is used to transmit to the missile the vertical error signal G_4. The signal is given to the automatic pilot at the same time as the "general pull-out pitch" command by means of a remote signal all or none (S_4).

- hydraulic locking of pitch-yaw control jacks,
- switching of the servo-control valves and putting the rotor into position on summation chains (1), (2), and (3),
- opening of the rotor hydraulic circuit,
- establishment of commands for collective and cyclic pitch at the input of summation chains (1), (2), and (3).

At the end of this sequence the rotor is brought under control (t = 1.5 sec).

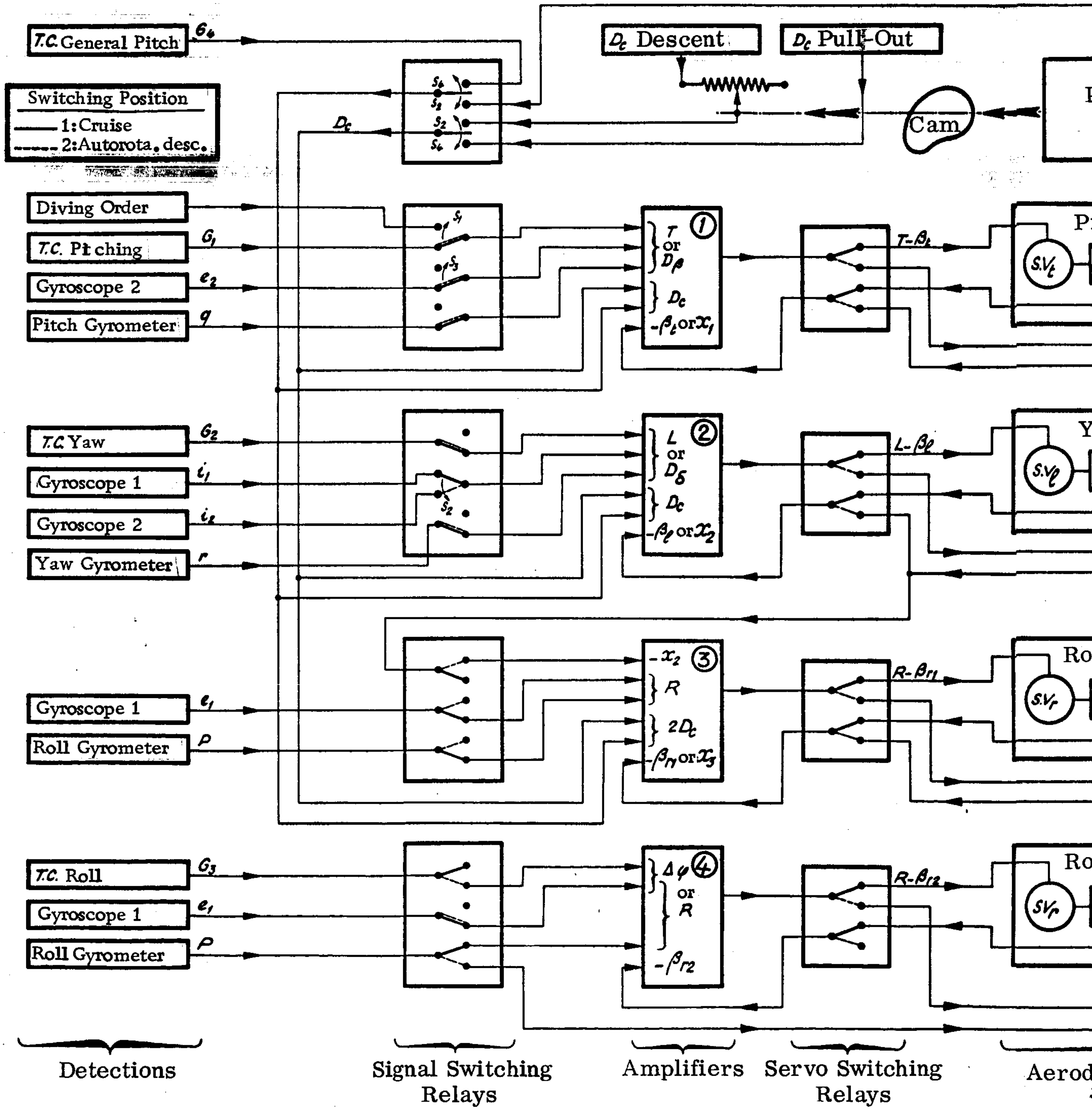
III - 3.3 - Culmination phase - accelerated descent

As soon as the rotor is activated, the cam of the programming motor delivers a progressive variation of the collective pitch permitting trim control by the rotor during the culmination phase and subsequent accelerated descent. This accelerated descent lasts approximately t = 13 sec. Thereafter the programming motor stops and delivers a constant general pitch adapted to the selected vertical descent speed (about 20 m/sec).

During this transitory maneuver (t = 1.5 to 13 sec) the remote command for pitch will deliver a trim program as a function of wind intensity (vertical trim at zero wind). Besides, the guidance commands for pitching and yawing will be progressively superposed on the stabilization terms.

III - 3.4 - Autorotative descent

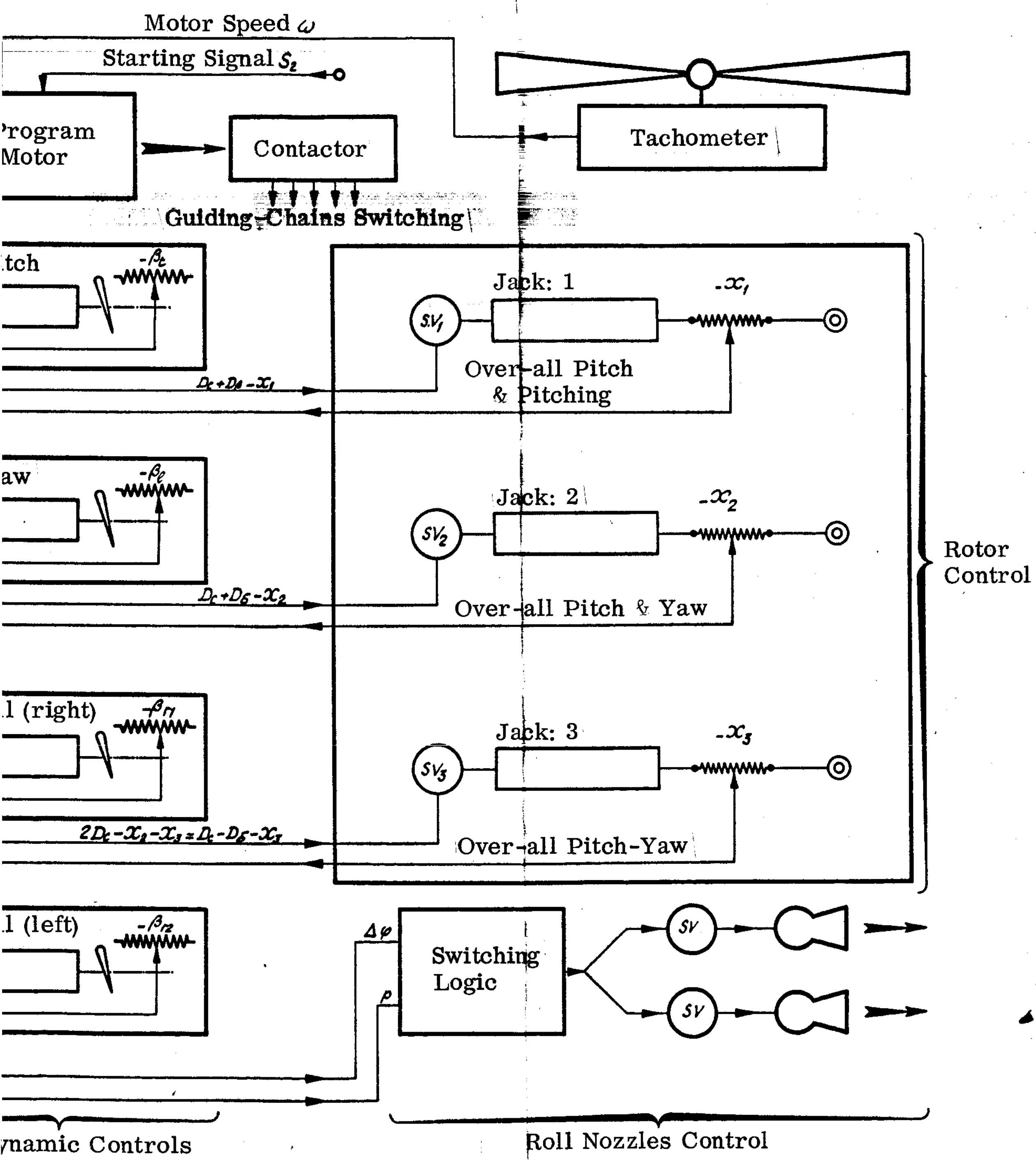
The automatic pilot receives continuously progressive remote signals for pitching and yawing (G_1 and G_2). These signals are guidance commands worked out on the ground as a function of missile deviations in meters, calculated as the horizontal projection on a fixed trihedron connected with the landing point. The commands are distributed between pitching and yawing channels as a function of rolling orientation that is commanded simultaneously by remote control to the missile. It should be noted that G_1 on the pitching channel results from the superposition of the guidance command on the balanced trim program. This program, which is established on the ground, delivers a constant correction during autorotative descent. In all cases superposition is achieved by the computer on the ground and a single "successive" remote control channel is used for pitching. The remote command all or none (S_3), transmitted when the desired altitude is reached, puts an end to the autorotative descent phase.



T.C. General Pitch
Switching Position
1: Cruise
2: Autorota. desc.
Dc Descent
Dc Pull-Out
Cam
Diving Order
T.C. Pitching
Gyroscope 2
Pitch Gyrometer
T.C. Yaw
Gyroscope 1
Gyroscope 2
Yaw Gyrometer
Gyroscope 1
Roll Gyrometer
T.C. Roll
Gyroscope 1
Roll Gyrometer
Detections
Signal Switching Relays
Amplifiers
Servo Switching Relays

123-1

Functional Diagram



APPENDIX IV

PATENTS ISSUED BY THE

"UNITED STATES PATENT OFFICE"

No. 2, 684, 213: Mixed type aircraft with fixed wing and rotary wing sustaining means (July 20, 1954)

No. 2, 886, 261: Remote control aircraft system (May 12, 1959)

Patented July 20, 1954 2,684,213

UNITED STATES PATENT OFFICE

2,684,213

MIXED TYPE AIRCRAFT WITH FIXED WING AND ROTARY WING SUSTAINING MEANS

Roger Aimé Robert, Boulogne-sur-Seine, and André Marcel Henault, St. Cloud, France; said Henault assignor to said Robert

Application August 14, 1947, Serial No. 768,626

Claims priority, application France September 10, 1946

24 Claims. (Cl. 244—3)

1

The present invention relates to a rocket-propelled flying machine or aircraft having great horizontal speed and capable of landing vertically at any given point.

Such a flying machine may be used, in particular, for carrying mail over long distances and at high speeds.

The aircraft according to the invention is characterized by the combination of a flying apparatus adapted for high speed horizontal flight with a rotary wing which is retracted during high-speed flight and whose axis of rotation is directed along the longitudinal axis of the aircraft, this wing allowing substantially vertical descent at very low speed upon landing.

The aircraft, as considered in its construction adapted for high-speed horizontal flight, may or may not be provided with sustaining wings; and it may or may not comprise stabilizing control wings arranged, if provided, either in front of or behind the sustaining wings.

It has already been proposed to use a flying apparatus constituted by a high speed plane provided with a rotary wing, whose axis of rotation is perpendicular to the longitudinal axis of the apparatus and the blades of which, in high speed flight, are folded back along the fuselage or are retracted therein.

This apparatus presents the following main drawbacks:

(*a*) The blades and their attachments are, at the time of their opening, submitted to great strain due to the relative wind especially if, at that time, the speed of the apparatus is great, which is in particular the case for a rocket-propelled machine.

(*b*) The area of the circle described by the tips of the blades is relatively small because the length of the blades is conditioned by the length of the fuselage of the apparatus and by the position of the axis of rotation of the rotor, as the latter must necessarily be located in the vicinity of the center of gravity.

(*c*) The unfolding of the blades, added to the necessity of keeping them away from the surfaces of the control wings during their rotation, creates complications.

In an aircraft according to the invention, since the axis of rotation of the rotor is arranged in the axis, or substantially in the axis, of the fuselage of the apparatus, the blades in their folded position are very readily lodged along the length of the body of the rocket.

The arrangement and the construction of the hub of the rotating wing are likewise very simple as this hub fits perfectly within the nose or forward outline of the fuselage.

2

In their open position and during their rotation, as the axis of the fuselage is then vertical, the blades have ample clearance with respect to the sustaining wings and the control surfaces, both of which may therefore be designed without having to take this rotor into account.

The rotor can have a maximum diameter since the length of the blades may be substantially equal to the length of the fuselage, while in the known machines mentioned hereabove, this length can be only half the length of the fuselage. In other words, the area swept by the rotor in the machine according to the invention may be four times as great as in a known machine of equal weight.

Finally, on landing, the blades of the machine according to the invention are not submitted to excessive stresses when they are unfolded.

Such an aircraft, when it approaches its destination, is directed, for instance by electromagnetic means, into an upright position, in order to bring it to a vertical climb, its propelling means being then advantageously cut off.

Due to its impetus, the aircraft continues its ascent substantially along a vertical path until its speed is annulled and then reverses. At this instant, that is when its speed is zero, or still rising but very low, or already dropping but very low, the unfolding or spreading of the blades of the rotor is effected. This manoeuver may be directly effected if there is a pilot on board, or through remote control, or automatically.

In any case, the stresses on the blades and their attachments remain very small.

The rotor is preferably caused to windmill by auto-rotation at the time of its spreading. This auto-rotation is caused either by the effect of a suitable incidence of the blades, or by the use of an auxiliary source of power.

Without departing from the scope of the invention, the rotor, instead of being free and operating by auto-rotation, may be driven by a source of power, similarly to the rotor of an helicopter, which makes it possible if desired still more to reduce the speed of the descent, or even to reduce it to zero.

The orientation of the axis of the rotor by inclination of its pivot, or by any other suitable arrangement, as for instance cyclic blade-pitch variation, makes it possible to generate at will an horizontal component which controls the descent of the aircraft along such a trajectory that it may be caused to land at any predetermined point, or at least a point determined with a good accuracy.

In its folded position, each of the blades of the rotor may be lodged in a channel formed along a generatrix of the fuselage, which channel may be covered by a fairing, or sealed by the blade itself.

When the blades present along their length a variation of incidence, they are lodged in channels or the like which, rather than extending along generatrices of the fuselage, extend along helicoidal curves traced on the latter.

In the following description, given as an example, reference is made to the accompanying drawings, in which:

Figure 1 is a view in elevation;

Figure 2 is a section along line 2—2 of Figure 1;

Figure 3 is an elevation showing the aircraft with its rotating wing unfolded during its windmilling fall;

Figure 4 is a view thereof on a larger scale, showing the means of attachment of the rotating wing;

Figure 5 shows the aircraft in various positions during the manoeuver preliminary to landing;

Figure 6 is an elevation of a modification; and

Figure 7 is a section along line 7—7 of Figure 6.

In the embodiment shown in Figures 1 to 5, the aircraft comprises a fuselage 10 on which is fixed a sustaining wing 11. At its rear end, the fuselage carries horizontal control fins 12 and vertical control fins 13. Inside the fuselage is housed an engine 10' which may be a reaction or jet engine. According to the invention, the aircraft further comprises a rotating wing 14, which is retracted in normal flight and is spread out just before the aircraft contacts ground so as to rotate about a longitudinal axis of the craft. This rotating wing is made up of a number of blades 15, three in the embodiment shown in Figures 1 to 5. During flight, the blades 15 are housed in corresponding grooves or channels 16 (Figure 4) provided in the surface of the body of the fuselage 10.

These grooves are disposed according to generatrices of the body of the fuselage as shown in Figure 1. If, however, the blades have an incidence variable along their length, the grooves are of helicoidal form, so as to insure proper housing for the blades and simultaneously to preserve the aerodynamic qualities of the aircraft.

Each blade 15 is rotatably mounted on a pin 16' carried by an arm 17 of the fuselage nose 37, the said nose or fair-shaped portion 37 and the arms 17, three in the example shown, being rigidly secured together by any convenient well-known means, such as angle-irons and bolts, etc. Near their point of pivoting, the blades 15 are each provided with a lateral arm 18 to the free ends of which are pivoted links or connecting rods 19, in turn pivoted at their other ends to a disc 20. Each of the connecting rods 19 is advantageously constituted by a shock-absorbing device. The latter is adjusted in such a way that, when fully extended, it forms a stop limiting the opening of the rotor and preventing hyper-extension of the blades upwards, while permitting the rotation of each blade around the corresponding pin 16'. Thus each blade can assume all the positions of equilibrium resulting from the combined action of the aerodynamic and centrifugal forces.

The disc 20 is carried by the rod 21 of a hydraulic jack 22 provided with universal joint means comprising, for example, a spherical swivel-surface 23. The latter cooperates with the spherical recess of a ring or spherical socket 24 having a cylindrical outer surface with end flanges 24' and 24'' and slidable in a complementary ring 25' carried by struts 25 rigid with the fuselage. The body of the jack 22 terminates in an upper tubular extension 22' which forms the pivot or inner race around which the rotating wing 14, comprised of the blades 15, may rotate through the medium of a ball bearing 26. The jack 22 may be subjected at its lower extremity to the action of two control jacks 27 and 28, at right angles to each other. These control jacks are pivoted as shown at 27' for the jack 27 on a ring 29' rigid with the body 22 and also pivoted as shown at 27'' at their outer ends to a transverse plate 29, resting on the fuselage 10 through the medium of a roller bearing 30, so as to be capable of rotating about the longitudinal axis of said fuselage, the outer race of the bearing 30 being rigid with the fuselage 10. This plate 29 is formed in its center with a recess 31, the shape of which corresponds to the shape of the lower extremity 32 of the jack 22. The recessed projection 33 of the plate 29, in which is formed this recess, forms the hub of a toothed gear 34 in mesh with the output pinion 35 of a motor 36 carried by the body of the fuselage 10.

During normal flight of the aircraft under the action of its propulsion means 10', the blades 15 are collapsed along the fuselage 10 (Figures 1 and 2) and are retracted within the contour of this fuselage so as to be flush therewith. Under the effect of the drag acting on the nose 37 of the rocket, the nose is applied against the top of the fuselage proper 10, the socket 24 abutting by its flange 24'' against the ring 25', which is the position for take-off, where the nose 37, including the blades 15, the arms 17 and the jack 22, rests under gravity on the fuselage proper 10, and the extremity 32 of the jack 22 is in the recess 31, thus maintaining the nose 37 strictly in the longitudinal axis of the fuselage 10 in spite of any transverse stresses which may act thereon. The cooperation of the projection 32 and the recess 31 has for its object to insure a positive alignment of the nose and fuselage during normal operation in opposition to bending stresses applied thereto. The nose and fuselage will thus be retained in positive alignment without resorting to the jacks 27 and 27' which would, in any event, have a certain inherent resilience.

The launching and adjustment of the controls of the aircraft are such as to cause the aircraft to reach the neighborhood of its destination. It is then brought, for instance by remote control, preferably automatic, to immediate proximity of its landing point. The landing manoeuver proper then starts. The rocket is guided so as to reach a point situated substantially vertically above the point where it is to land with its nose turned toward the zenith, at a speed close to zero. For instance, the trajectory of the rocket (Figure 5), which is first substantially horizontal, is deflected upwards (position I), the speed of the rocket diminishing progressively until it becomes substantially zero when it reaches the vertical position (position II). By remote control or automatically, the rotating wing 14 is then unfolded through the action of the jack 22, which draws the disc 20 downwards from the position shown in dotted lines to the position shown in solid lines (Figure 4). During this unfolding or spreading movement, which is braked by the action of the shock absorbers forming the connecting rods 19, neither the rotating wing, nor its attachments are subjected to excessive strain, the speed of the rocket being then substantially nil. Under the action of gravity, the aircraft drops substantially vertically, its fall being slowed down by the rotating wing 14, then in rotation. Due to the length of the blades 15, which is substantially equal to the overall length of the aircraft, the speed of this fall is very low.

The rotating wing may be automatically put into rotation under the effect of a suitable incidence of the blades 15.

In a modification, the rotating wing is put into rotation by means of an auxiliary source of energy.

During this fall, the body of fuselage proper 10 hangs from the nose 37 carrying the windmilling rotary wing, and said nose thus tends to be slightly spaced from the fuselage proper 10, as clearly shown Figure 5 in position II, the extremity 32 of the jack 22 being pulled out of the cavity 31; in this movement, the socket 24 slides in the ring 25′ until abutment against the flange 24′.

The retarded fall of the aircraft is guided from the ground automatically by remote control or manually from the aircraft so that the aircraft will contact the ground accurately at the desired point, for instance the roof of a post-office (position III).

This guiding is carried out by suitable inclinations of the axis of rotation of the wing 14 relatively to the vertical axis with which the fuselage 10 remains coincident. These inclinations are controlled through the control jacks 27 and 28.

To enable such guiding control to be effected from a remote station outside the aircraft, as for instance from the precise alighting location, the invention provides means, well-known per se, such as those disclosed in United States Patents 2,450,071 and 2,454,022, for controlling at the start of the rocket's fall, the direction of said jacks with respect to a fixed (geographic or magnetic) bearing line at the landing, for instance by bringing the axis of one of said control jacks along said bearing line. These means comprise the motor 36 which is actuated for rotating the plate 29 on the adequate angle.

Figures 6 and 7 show a modification according to which the blades 38 of the rotating wing are two in number and, in their collapsed position, are housed edgewise in the body 10 of the fuselage.

We claim:

1. In a flying machine: a forwardly-truncated fuselage, a rotary nose at the forward end of said fuselage having its axis of rotation substantially coincident with the longitudinal axis of said fuselage, a rotary wing supported by said nose, said rotary wing comprising blades long enough to ensure that said machine will descend with its fuselage substantially vertical at a landing speed low enough for a proper landing, and means for retracting said blades by folding said blades down along said fuselage and nose.

2. In a flying machine as in claim 1, channels formed substantially longitudinally along the outer surface of said fuselage to house said blades.

3. In a flying machine as in claim 1, recesses formed substantially longitudinally on the outer surface of said fuselage to house said blades with the outer surfaces of said blades merging substantially flush with the outer surface of said fuselage.

4. A flying machine comprising a fuselage, a longitudinally-extending rotative hub in said fuselage forwardly thereof, means for supporting said hub slidably in said fuselage along a longitudinal direction thereof between a remote position and proximate position, blades forming a rotary wing structure supported on said hub, means for maintaining said blade along said fuselage in retracted condition, means for locking said hub with the axis thereof along the longitudinal axis of said fuselage when said hub is in proximate position, said last-named means being arranged to be disengaged by movement of said hub to the remote position for freeing said hub for transverse movement.

5. A flying machine comprising a forwardly-truncated fuselage, a jack at the front end of said fuselage, said jack having a body, a spherical swivel surface on said jack body, means for supporting said jack body from said fuselage through said spherical swivel surface, a wing structure rotatably mounted on said jack body, said structure including blades, a nose extending from said fuselage at the forward end thereof, means for connecting said blades with said nose for folding retraction of said blades along said fuselage towards the rear end thereof and opening expansion of said blades to form a rotary-wing structure transverse to said fuselage, said connecting means comprising a movable jack-element cooperating with said jack body, and a pivotal linkage between said blades and said movable element.

6. A flying machine as in claim 5 wherein said pivotal linkage comprises damping means.

7. In a flying machine as in claim 5, a transverse plate at the forward end of said fuselage, a central recess in said plate, and a projection at the lower end of said jack for a removable cooperation with said recess.

8. In a flying machine as in claim 5, a transverse plate forwardly of said fuselage, a central recess in said plate, a projection at the bottom end of said jack removably cooperating with said recess, means for rotating said plate around the fuselage axis by a predetermined angle, an angularly adjusting jack between the jack body and said plate, and a further angularly adjusting jack at right angle with the first-mentioned adjusting jack interposed between said jack body and said plate.

9. In a flying machine: a fuselage, a structure mounted at the forward end of said fuselage and rotatable relatively thereto, and a rotary wing carried by said structure transverse to said fuselage and adapted to limit the speed of descent of said machine with its fuselage substantially vertical to a value admitting of a proper landing and comprising blades foldable along said fuselage during horizontal flight of said machine.

10. In a flying machine: a forwardly truncated fuselage, a fin structure on and propulsion means in said fuselage, a rotary section extending forwardly from said fuselage, rotary wing blades on said rotary section, and means for bringing said blades from an operative position transverse to the fuselage longitudinal axis to an inoperative position substantially parallel to said axis and retracted along the fuselage and vice-versa.

11. In a flying machine: a forwardly-truncated fuselage, fin means and propulsion means respectively on and in said fuselage, a rotary nose extending forwardly from said fuselage, rotary wing blades on said nose, means for shifting said blades between an operative position transverse to the fuselage longitudinal axis, in which said blades exert a parachute action, and an inoperative posi-

tion substantially parallel to said axis in which said blades are retracted along said fuselage, and means for adjusting the longitudinal axis of said nose relatively to said longitudinal axis of said fuselage to control the direction of the parachutal descent of said machine.

12. Flying machine as in claim 11, wherein said nose is mounted for universal swivel movement in a support rigid with said fuselage.

13. Flying machine as in claim 12, including a swivel mounting rigid with said nose and mounted for sliding movement therewith parallel to the longitudinal fuselage axis relatively to a support rigid with said fuselage.

14. Flying machine as in claim 11, wherein said nose is supported by a swivel device mounted for longitudinal sliding movement relatively to the fuselage and slidable to and from a remote first position and a second proximate position relatively to said fuselage, and means for centering said nose relatively to said fuselage in said second position of said nose.

15. A flying machine as in claim 11 wherein said adjusting means comprise a swivel device supporting said nose, and a jack means for controlling the angle of the longitudinal axis of said nose relatively to the longitudinal axis of said fuselage.

16. In a flying machine: an elongated forwardly truncated fuselage, a tailfin structure carried at the rear part of said fuselage, a nose extending forwardly from said fuselage and rotatively mounted with respect to said fuselage, a rotary wing carried by said nose and including blades and means for folding said blades back along the fuselage towards the rear end thereof and for expanding said blades to their operative position transverse to the longitudinal axis of said nose.

17. In a conventional airplane including a fuselage with a sustaining surface and a tail-fin structure, in combination: a nose rotatively mounted on the fuselage forwardly thereof, rotary wing-blades carried by said nose and retractable along said fuselage substantially over the whole length thereof, and means for expanding said blades to constitute a rotary wing structure transverse to said fuselage for parachute descent on landing.

18. Aircraft comprising an elongated fuselage, a tapering nose section supported by said fuselage forwardly thereof for rotation relatively to said fuselage about a substantially longitudinal axis, rotary wing blades supported by said nose section and angularly equi-spaced about said longitudinal axis, and linkage means for retracting said blades along said nose and said fuselage and for expanding said blades transversely of said longitudinal axis.

19. Aircraft as in claim 18 comprising: universal connecting means between said nose and said fuselage; and means for adjusting the angular inclination of the nose axis with respect to the fuselage axis.

20. Aircraft as in claim 18, wherein the fuselage comprises aerodynamic stabilizer fin means.

21. Aircraft comprising: an elongated fuselage, universally mounted pivot means forward of said fuselage, means for controlling the inclination of the axis of said pivot with respect to the longitudinal axis of said fuselage, means for supporting said pivot means on said fuselage for sliding movement relatively thereto, stop means between said fuselage and said pivot means for limiting said sliding movement, a tapering nose supported on said pivot means, and substantially forming an extension of said fuselage, rotary wing blades supported on said nose and angularly equispaced about the centre axis thereof, means for expanding said blades and for retracting them along said nose and said fuselage, centering means respectively supported on said fuselage and on said nose and operative when said fuselage and said nose are in their adjacent condition and inoperative when said fuselage and said nose are in their spaced apart condition.

22. A flying machine comprising a fuselage, a nose extending forwardly from said fuselage, universal joint means supporting said nose on said fuselage, a rotary wing-structure supported on said nose and regularly arranged around the axis thereof, means for supporting said nose for lengthwise sliding movement with respect to said fuselage between a remote and a proximate position with respect thereto, means for maintaining said nose with the longitudinal axis thereof in alignment with the longitudinal axis of said fuselage when said nose is in said proximate position and inoperative when said nose is in its said remote position, and means for retracting said rotary wing structure along said nose and fuselage.

23. A flying machine which comprises: a forwardly-truncated fuselage, propelling means in said fuselage, a nose carried at the forward end of said fuselage and rotatable relatively thereto, and a rotary wing structure adapted for windmill action mounted on said nose and comprising blades and means for bringing said blades to an operative position transverse to the axis of said fuselage and for bringing said blades to an inoperative position retracted in folded condition along said fuselage.

24. A flying machine which comprises: a fuselage, a fin structure secured to said fuselage, a nose carried at the forward end of said fuselage, a rotary wing structure carried by said nose including blades, and means for folding each blade along the fuselage by rotation about an axis transverse to the long axis of said fuselage to render said rotary wing structure inoperative for horizontal flight of said machine and for expanding said wing structure by a reverse rotation of each one of said blades about said transverse axis to render said rotary wing structure operative effectively to slow down the rate of descent of said machine with its fuselage substantially vertical for landing.

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July 20, 1954 R. A. ROBERT ET AL 2,684,213

MIXED TYPE AIRCRAFT WITH FIXED WING AND ROTARY WING SUSTAINING MEANS

Filed Aug. 14, 1947 2 Sheets-Sheet 1

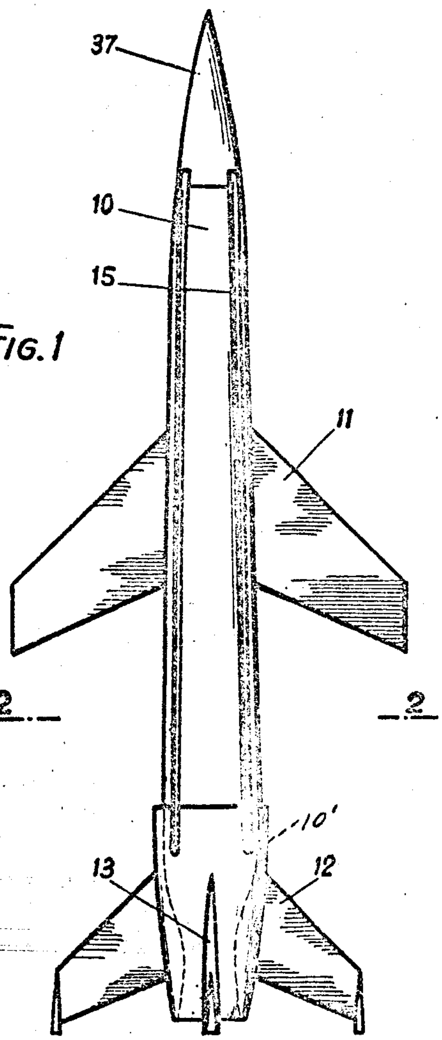

FIG. 1

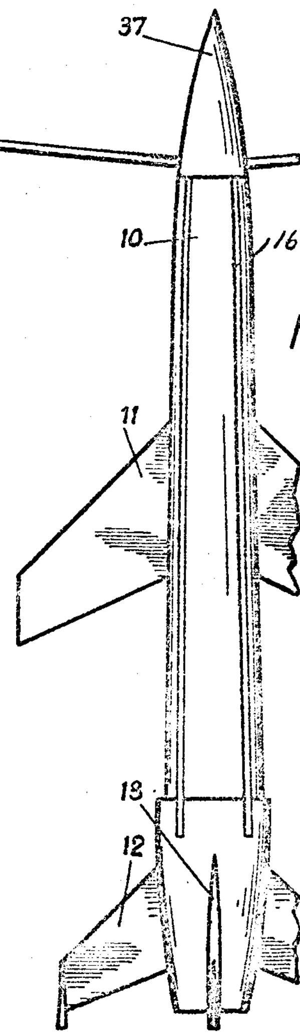

FIG. 3

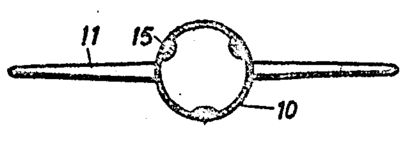

FIG. 2

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MIXED TYPE AIRCRAFT WITH FIXED WING AND ROTARY WING SUSTAINING MEANS

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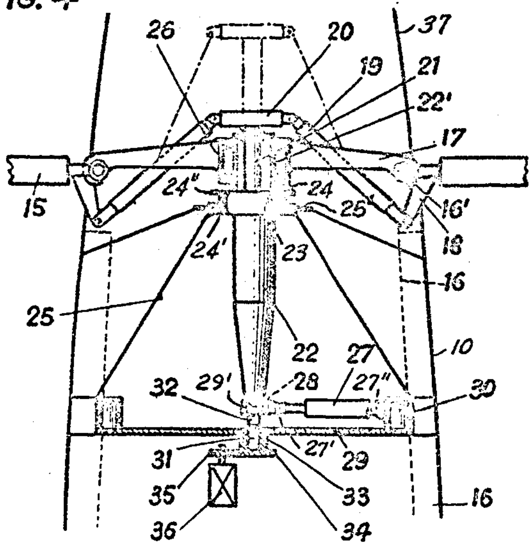
FIG. 4



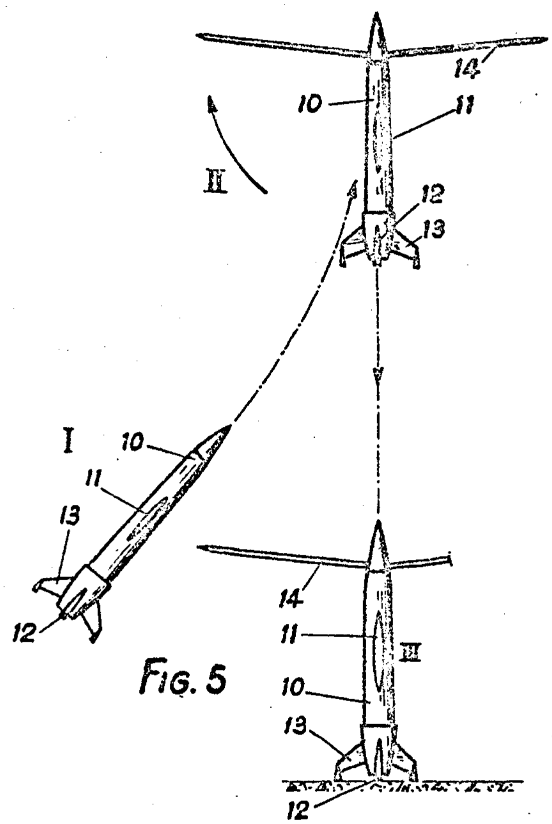


FIG. 5

FIG. 6

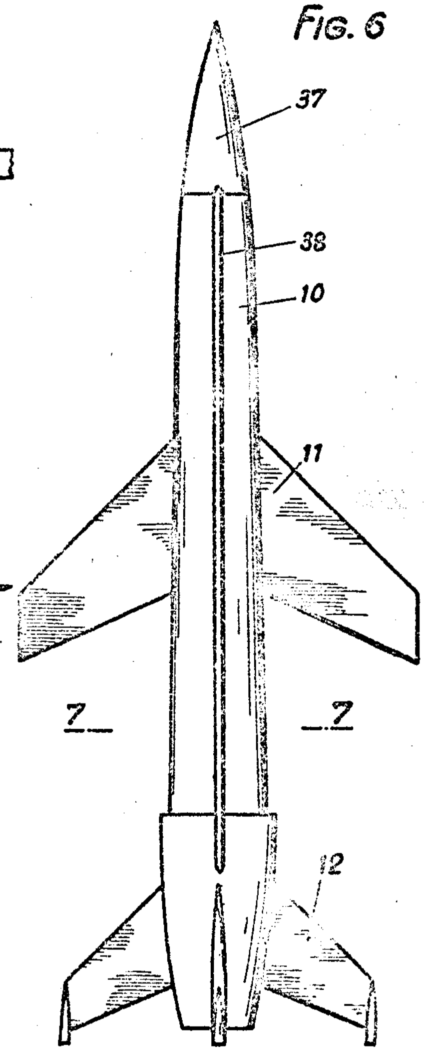
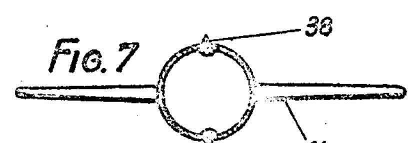


FIG. 7



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2,886,261

REMOTE CONTROL AIRCRAFT SYSTEM

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Application May 25, 1954, Serial No. 432,169

6 Claims. (Cl. 244—17.25)

This is a continuation-in-part of our Patent No. 2,684,213.

This invention relates to craft guidance means, and more particularly to apparatus for guiding a rotary-wing aircraft towards a predetermined landing point.

It is one object of the invention to provide apparatus whereby an observer stationed on the ground at or near a desired landing location will be able to guide the autorotational descent of a rotary-wing aircraft accurately towards said location, even in the absence of any human pilot aboard the craft.

According to the invention, a reference direction passing through the desired landing point or a point adjacent thereto is established aboard the aircraft to serve as a reference for the actuation of automatic means on the craft operative to modify the path followed thereby during landing.

The said reference direction characteristically associated with the landing point may be magnetic North or geographic North or any other suitable direction. The reference direction is reproduced on board the craft through radio transmission.

The invention is desirably applied to an aircraft the rotary-wing structure of which is carried on a hub universally mounted on the craft structure through a swivel joint or the like whereby the tilt of the rotary wing axis relative to vertical may be varied in any desired direction within the limits of a solid cone in order to direct the landing path of the craft.

In one embodiment of the invention as applied to a craft of the type just specified, the said hub is angularly positioned by the combined action of a pair of actuators—e.g. hydraulic motors or jacks—arranged out of alignment with each other, and preferably at right angles with each other, carried on a rotary support or platform perpendicular in extent to the normally upstanding axis of the rotary-wing structure, means being provided for rotating said support from the ground whereby a given direction of said platform may be set parallel to the reference direction.

An embodiment of the invention will now be described by way of illustration with reference to the accompanying diagrammatic drawings wherein:

Fig. 1 is a perspective view showing part of the rotary-wing structure and the actuating means therefor according to the invention;

Fig. 2 is a block diagram further illustrating a fragment of the structure of Fig. 1, in axial section on an enlarged scale;

Fig. 3 is a block diagram of the ground control station.

As shown in Fig. 1, a rotary-wing structure comprises a hub 10 and wing blades 11 carried thereon, the hub 10 being provided with a spherical swivel portion 12 adapted to cooperate with a complementary spherical socket 13 forming part of the fixed frame of the aircraft.

Coupled to the lower end of the hub 10 through a ball 14 are the displaceable elements 15 and 16 of a pair of actuators or jacks 17 and 18 comprising cylinders 19 and 20. The displaceable elements 15 and 16 may constitute piston rods attached to pistons 21 and 22 slidable in the cylinders and defining therein the chambers 23—24 and 25—26 respectively. The actuator cylinders 17 and 18 have clevises 27 and 28 formed on the outer ends thereof by means of which the cylinders are pivoted to lugs 27′ and 28′ projecting from the flange 29 of a rotatable platform 30 comprising an annular peripheral channel element providing a runway cooperating with a plurality of horizontal rollers 31 rotating on brackets 33 secured to the frame 32 of the aircraft.

The platform 30 (Fig. 2) is provided with a central boss or hub 34 in which a recess 35 is formed adapted to cooperate with an appendage 36 forming the lower end of the hub 10 for centering the latter when the craft is not in condition for an automatic landing according to the invention. Formed around the periphery of boss 34 is a gear 36 adapted to mesh with a pinion 37 secured on the output shaft 38 of a suitable electric motor 39. The motor 39 is energized for selective rotation in opposite directions through leads 41 and 42 from the output of a suitable radio receiver 40 of conventional design herein illustrated in block form. The actuators 17 and 18 are controlled through a radio receiver 43 mounted on the platform 30 and adapted in response to control signals received thereby to operate either or both of the actuators selectively so as to cause the displaceable members to be moved into or out of the related cylinders. Both the receiver 40 and the receiver 43 are adapted to receive control signals from a remote station P at or near the predetermined landing location. The control station P comprises means for establishing a directional reference, for example a magnetic compass 44 or equivalent means, and a radio transmitter 45 having an aerial 46 for transmitting radio signals capable of being received by the receiver 40 and converted thereby into suitable control signals fed to the reversible motor 39 to rotate the latter in either direction depending on the polarity of phase condition of the control signals. Rotation of the motor 39 rotates the platform 30 in a corresponding direction until a predetermined direction or radius of the platform has been brought into alignment with reference direction as indicated for example by the index 47 of the compass. Any suitable servo-system may be used for operating the motor 39 in the manner just described. Also provided at the control station P is a further radio transmitter 48 provided with two suitable adjusting knobs or the like 49 and 50 whereby radio signals may be transmitted to the receiver 43 for operating the actuators 17 and 18. For example, the adjusting knob 49 may serve to control the position of the actuator 17, the displaceable element of this actuator being moved inwards or outwards according as the knob 49 is moved to one or the other side of a neutral position. Similarly the knob 50 will serve to control the position of the actuator 18.

The arrangement described operates as follows: When the ground operator located at the station P catches sight of the aircraft which previously was controlled in some other suitable manner, and as the craft commences its autorotational descent, the path of the craft is guided towards the predetermined landing point in the following way: The switch 51 of transmitter 45 is placed in operative position. Transmitter 45 constitutes the transmitter unit of a radio remote control system whereof the receiver unit is provided by the receiver 40. As disclosed above the system operates to bring a predetermined radial direction of platform 30 into, and maintain it in, alignment with the reference direction established by the directional instrument 44. This establishes a directional reference on board the craft, said reference remaining fixed regardless of the craft's position and attitude, and the operator then adjust the knobs 49 and 50 to guide the descent of the craft as easily and accurately as if he were on board. That is, by observing the position of the craft the operator is able at any instant to correct any displacement the craft may tend to assume off the ideal landing path towards the preselected point, and thus bring in the craft safely and surely to a landing at said point.

What we claim is:

1. A guidance system for a rotary-wing aircraft for guiding the craft in descent toward a preselected landing area, which comprises means for reproducing on said aircraft a predetermined reference direction at said area, and means for controlling the tilt of the axis of rotation of the rotary wing structure of said aircraft with reference to said fixed direction as reproduced on said craft.

2. A system for guiding a rotary-wing aircraft in descent towards a preselected landing area, comprising, a remote control transmitter station, signal transmitting means at said station for transmitting signals indicative of a predetermined topographical direction at said landing area, means on said craft for receiving said signals and for maintaining at all times a selected direction of the craft in a constant predetermined angular relationship with the said topographical direction, means on the craft for controlling the tilt of the axis of rotation of the rotary wing structure thereof, and further transmitter means on said landing area for remotely controlling said tilt control means.

3. A system for guiding a rotary-wing aircraft from a preselected landing area, which comprises, a rotary-wing structure on said craft having a hub, means mounting said hub for universal rotation on the craft, a support mounted for rotation about a predetermined axis of said craft, a first actuator mounted on said support and operatively connected with said hub, a second actuator mounted on said support at an angle to said first actuator means and operatively connected with said hub, first radio receiver means for actuating each of said actuator means in either one of two opposite directions, second radio receiver means and means responsive thereto for adjusting said support angularly about said predetermined axis, transmitter means at said area constituting the transmitter of a remote control system cooperating with said second receiver for reproducing a topographical direction of said area as a related angular position of said support, and further transmitter means at said area adapted for cooperation with said first receiver to actuate said actuator means.

4. In a rotary wing aircraft having a shaft for the rotary wing, means mounting the shaft for universal rotation and jack means for controlling the relative vertical tilt of the shaft: a support for the jack means mounted for rotation around the shaft axis with said axis vertical, and means for controlling the rotation of the support from a control station outside the aircraft.

5. In a rotary wing aircraft comprising a wing shaft mounted for universal rotation and two jacks arranged at an angle to each other and operatively connected to the shaft to vary the angle thereof with respect to a mean position: a platform perpendicular to the shaft in said mean position and mounted for rotation on the aircraft about said shaft, drive means for rotating the platform in either direction, means for controlling the drive means from a control station outside the aircraft, and remote control means for actuating either jack for extension or retraction.

6. A remote-control system for landing a rotary wing aircraft having a universally rotatable wing shaft and comprising two angularly disposed jacks each having one end operatively connected to the shaft to vary the inclination thereof with respect to a mean vertical position, said system comprising: a platform occupying a position transverse to said shaft in the mean vertical position thereof and mounted on the aircraft for rotation about the axis of said shaft, the two jacks having their other ends supported by said platform, drive means for rotating the platform, radio control means for the drive means comprising a first receiver on the aircraft and a first transmitter at the landing site, actuating means for extending and retracting each of the jacks selectively, and radio-control means for the actuating means comprising a second receiver on the aircraft and a second transmitter at the landing site, said second transmitter comprising two actuating members corresponding respectively to the two jacks for extension or retraction thereof in accordance with actuating movements imparted to the respective member in either direction relatively to a mean position corresponding to the said mean vertical position of the wing shaft.

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REMOTE CONTROL AIRCRAFT SYSTEM

Filed May 25, 1954

Fig.1

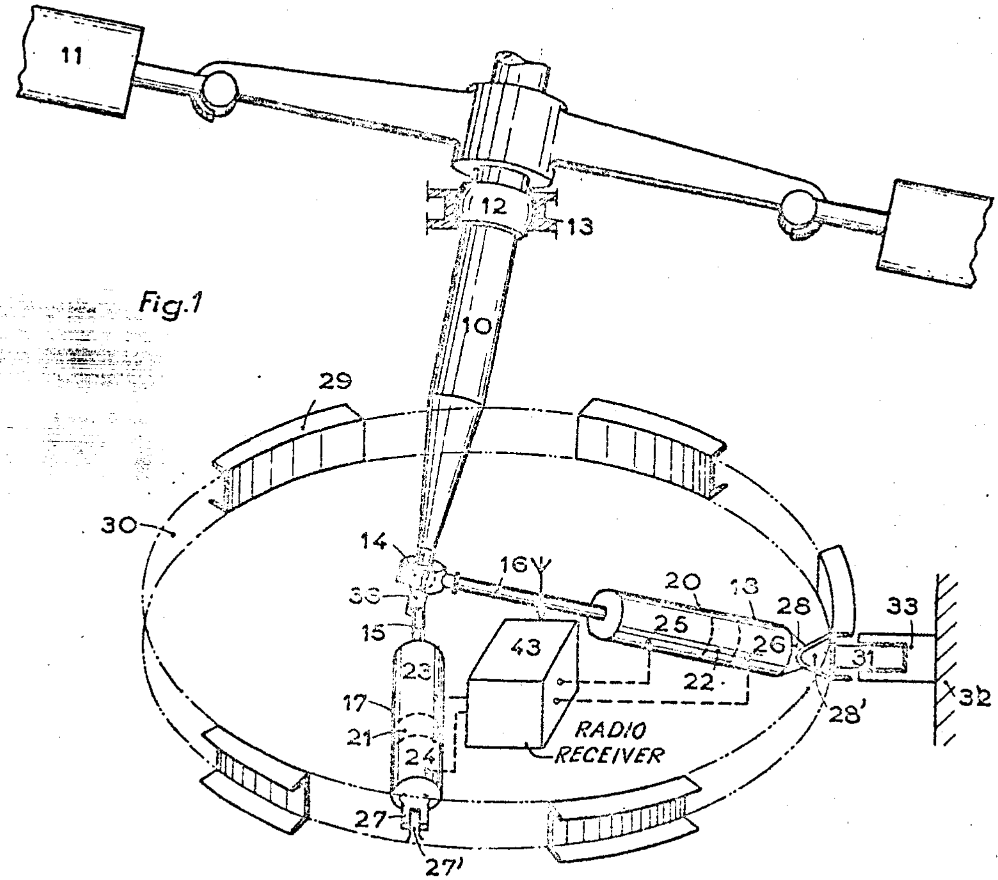
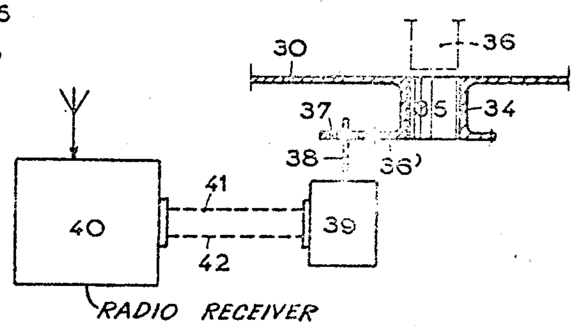


Fig.2



P

Fig. 3